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LUPA: An Excursion Vehicle for the Moons of Mars

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AIAA Foundation Undergraduate Space Mission Design Competition

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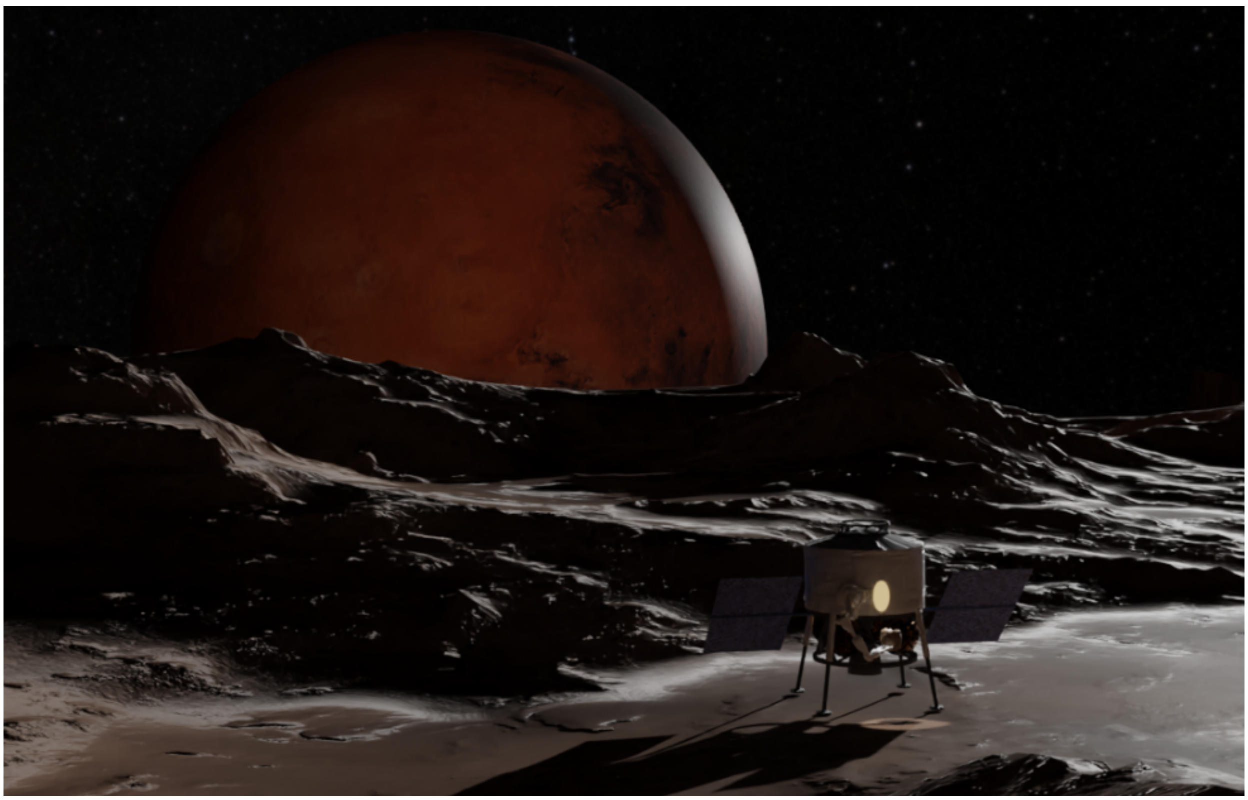

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List of Abbreviations

C_3, C3 – Orbital characteristic energy
DST – Deep Space Transport
EEV – Exploration Excursion Vehicle
ΔV, dV, deltaV – Change in velocity
IDR – Initial Design Review
LH2 – Liquid hydrogen

LOX – Liquid oxygen
NTO – Dinitrogen tetroxide
RP1 – Rocket-Propellant 1
UDMH – Unsymmetrical dimethylhydrazine
RDT&E - Research, development, test, and evaluation

1 Overview

1.1 Design Requirements and Constraints

The key design requirements and constraints of this project as defined in the official AIAA competition document [1] are as follows. The original list of requirements as they appear in the competition document are located in

Table 18 in the Appendix.

1.2 LUPA Mission Profile

LUPA's primary mission, whose comprising parts are each discussed in greater detail within this report, is to serve as a short-duration excursion vehicle capable of performing all necessary orbital maneuvers involved in autonomous rendezvous with a Deep Space Transport (DST) vehicle carrying crew and with the two largely unexplored moons of Mars, Phobos and Deimos, collecting surface samples and performing scientific experiments at each destination.

Launching in 2035, most of LUPA's operational lifespan will be spent vacant in orbit of Mars occasionally performing stationkeeping maneuvers, performing system health diagnostics, or otherwise serving as a relay point in our fledgling interplanetary infrastructure. On 1 January 2040, after nearly 4 years in Martian orbit, crew arriving aboard a DST vehicle will enter a 5-sol parking orbit with which LUPA will autonomously rendezvous, marking the beginning of the designed 15-day excursion from the DST to the moons of Mars.

At the end of LUPA's excursion, all crew, samples, and useful equipment will be transferred to the DST and LUPA will be left to rest in space having fulfilled her purpose.

2 Trajectory

2.1 Launch Vehicle Selection

Per the project guidelines, the total cost of the vehicle is not to exceed $1 billion USD. If you're trying to get to Mars on a budget, the clear choice is to do it on the SpaceX Falcon Heavy. While the amount of deltaV needed to put payload on a Mars-intercept trajectory requires too much propellant for the Falcon stages to be recoverable, the recoverable booster model employed by SpaceX has brought the cost per kilogram to orbit down significantly and their vehicle is currently the cheapest lift vehicle capable of sending payload to Mars.

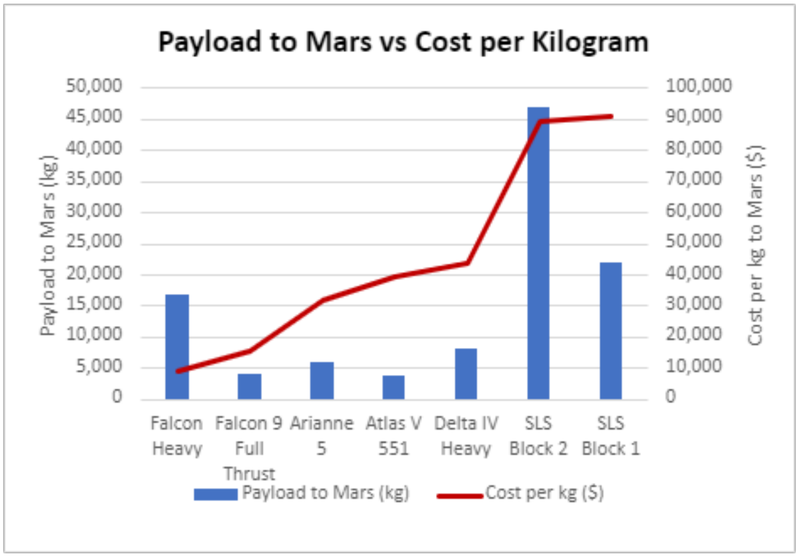


Figure 1: Launch vehicle cost and capability comparison

The tradeoff of selecting the Falcon Heavy over other Mars-capable heavy lift vehicles is the reduced payload to Mars capability. Following the proposed development timeline outlined in section ADD SECTION, an additional 14 years exist between the time of this report being written and the launch of LUPA. This is certainly enough time for alternative launch options which are presently unavailable, such as NASA's SLS lift vehicle, to be considered as candidates for our vehicle. However, as can be clearly seen in Figure 1, the cost per kilogram skyrockets after the Falcon Heavy. Every additional kilogram sent to orbit corresponds to energy that will need to be managed upon arrival at Mars. For this reason, the comparatively limited launch capability of the Falcon Heavy is overshadowed by its cost benefit, leading to its selection as our launch vehicle.

Since SpaceX is a commercial launch provider, an additional benefit of selecting the Falcon Heavy is the availability of customer resources about the vehicle. Of particular use to this project was the dimensions of the payload fairing included in their user's guide [2].

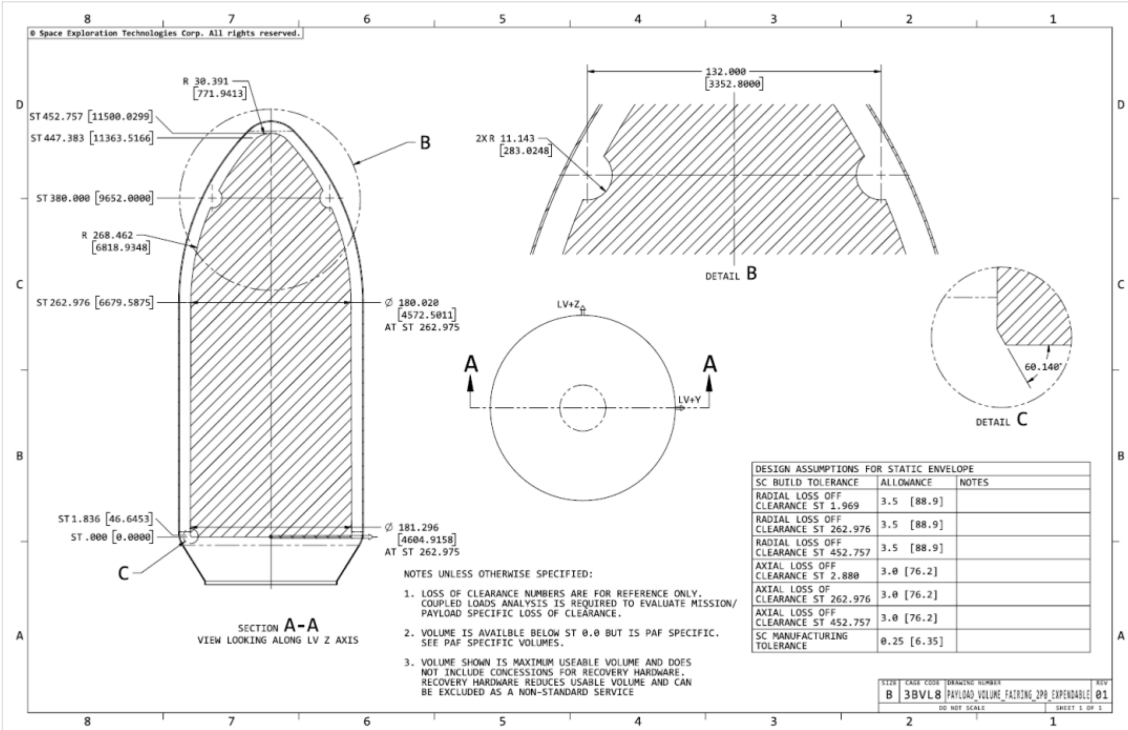


Figure 2: Engineering drawing of the Falcon 9 payload fairing with relevant dimensions

2.2 Launch Window Determination

It is common knowledge that the relative orbits of Earth and Mars are such that, roughly every 18 months, a period of time exists where the opportunity to send payload from one to the other is advantageous from both a time and energy perspective. During these periodic windows, the travel time, marked by the time between departure from Earth and arrival at Mars, is anywhere from 200 to 400 days. Per the design constraints of the AIAA competition, LUPA needed to be ready and waiting in a 5-sol orbit by 1 January 2040. Extrapolating backwards points to the latest launch date occurring during the window that opens in September 2037. However, as can be seen in Figure 3 below, this September 2037 window is one of the more significantly expensive ones from the perspective of both time and characteristic energy upon arrival.

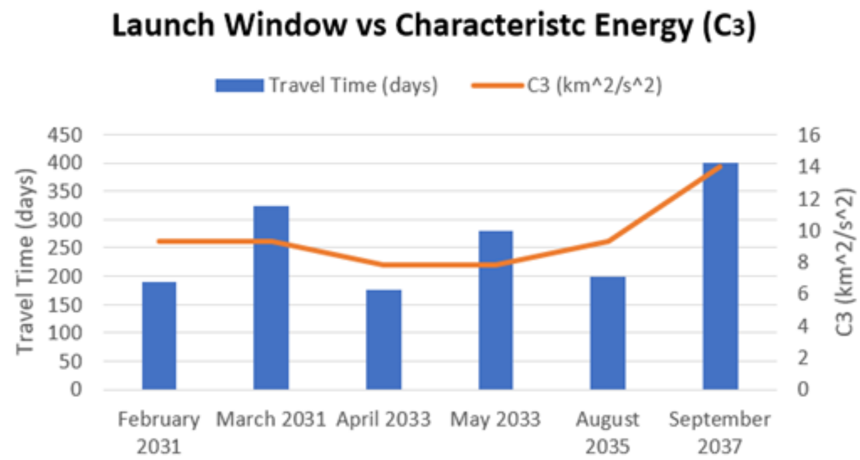


Figure 3: Launch windows during the 2030s

The most efficient launch window occurring during the 2030s occurs in April 2033, however choosing this window for launch would mean that LUPA would have to remain in space for the better part of a decade. Naturally, the August 2035 window was selected due to its relatively standard time of flight and considerably more favorable energy requirements. As discussed in section 2.1, Launch Vehicle Selection, keeping mass low was a key design consideration. The greater the characteristic energy upon arrival at Mars, the more propellant will need to be expended in order to perform orbital injection and thus the mass of the vehicle increases proportionally.

The determination of a more specific launch date is discussed further in the beginning of section 9.1, ΔV Requirements, due to the direct relationship between characteristic energy upon arrival and the subsequent ΔV and propellant mass requirements.

3 Sizing Calculations

Due to the selection of the SpaceX Falcon Heavy as the lift vehicle, the maximum geometric dimensions and gross weight were governed, respectively, by the dimensions of the payload fairing (See Figure 2) and the maximum rated payload mass to Mars of 16,800 kilograms (37,037 pounds).

Another key dimension was the habitable volume requirements. As LUPA will be a crewed vehicle, the physiological needs of crew must be taken into account. This most significantly amounts to the definition of how much volume must exist within the vehicle in order for two adult humans to comfortably carry out mission operations. A 2011 paper published by NASA addressed this exact problem [3]. In their research, a plot was created which indicated a logarithmic trend in the necessary habitable volume per crew member per mission duration in days, seen below in Figure 4.

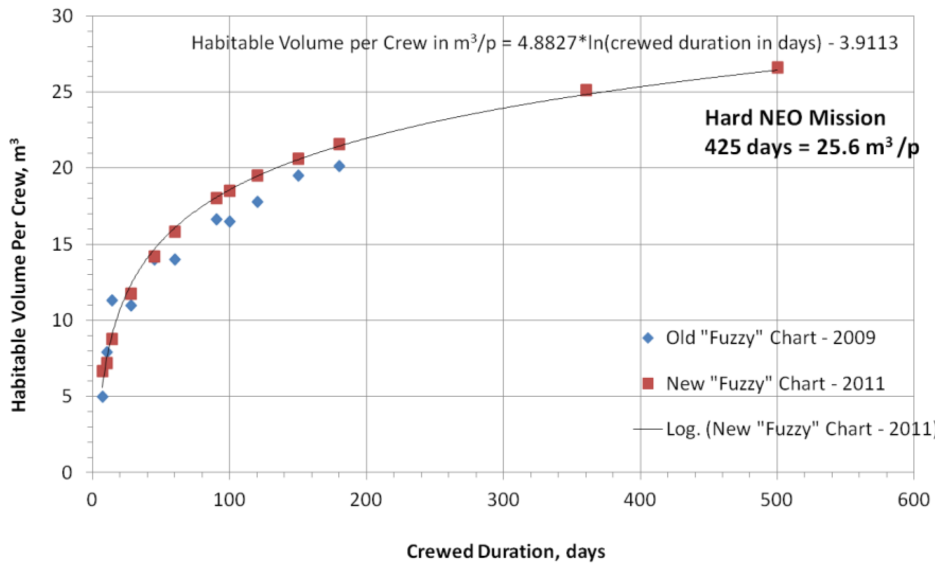


Figure 4: Averaged habitable volume curve

Initially, LUPA's mission was designed to be a single, 30-day sortie. However, it was revealed through our calculations that the mass associated with the volume required by a 30-day mission for two crewmates went well above our mass budget which was largely governed by the aforementioned payload capacity of the Falcon Heavy in addition to the mass associated with the propellant necessary to carry out the necessary orbital maneuvers. Ultimately, the sortie length was cut in half to 15 days, corresponding to a minimum habitable volume of 18.55 cubic meters (655.09 cubic feet).

With the volume and maximum allowable diameter known, the height of the vehicle was trivial to calculate using the equation for the volume of a cylinder. Figure 5 showcases how the overall size and configuration of the vehicle changed as mass and volume calculations were iterated upon, with the final configuration being on the far right.

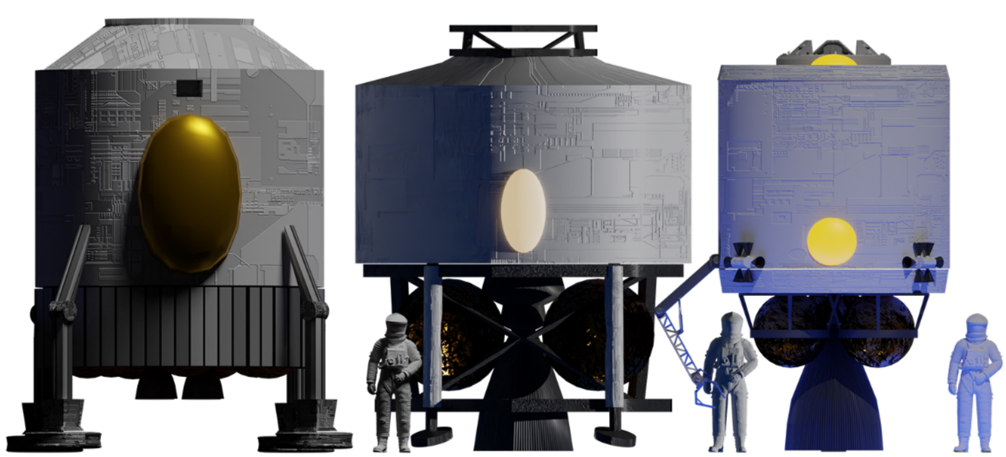


Figure 5: LUPA size iteration

The final configuration of LUPA featured an internal pressure vessel of diameter of 3.074 meters (10.085 feet) and height of 2.750 meters (9.022 feet).

4 Science Objectives

4.1.1 Internal Science Equipment

While a majority of this mission's scientific equipment will be external but there is still a portion that will be internal and accessible to the astronauts. To reduce the launch weight of LUPA the science equipment will be delivered to LUPA aboard the Deep Space Transport along with the astronauts. The scientific equipment, quantity, and mass are shown in **Table 3**.

Table 1: Internal Scientific Equipment Quantity and Mass

Item	Quantity	Weight (kg)
HP Zbook 15 Mobile Workstation	5	11.748
PNY 32GB Flash Drive USB 2.0 Type A	50	0.050
Nikon D6	2	2.540
AF-S NIKKOR 800mm f/5.6E FL ED VR	2	9.180
AF-S NIKKOR 24-70mm f/2.8E ED VR	2	2.140
AF-S Teleconverter TC-14E III	2	0.380
IX Cameras i-SPEED 727	1	8.482
Cleartech Compact Vacuum Glove Box	1	49.895
Vacuum Pump	1	12.247
Vacuum Control Unit 120V	1	9.072
Vacuum Tubing, 10FT	1	0.860
Resonon Pika NIR-640 Hyperspectral Camera	1	3.210
Resonon Pika XC2 Hyperspectral Camera	1	2.570
Resonon Outdoor Field System	2	47.000

The inclusion of the HP Zbooks is to manage systems and equipment across the vehicle while the flash drives are for data storage and data transfers between systems and laptops. The function of the Nikon cameras and it's lenses is for astronaut directed and unplanned photography. The IX Camera slow-motion camera is to study the motion of the soil of Phobos and Deimos during sample collection. The vacuum glove box is for the manipulation of certain collected samples while the pump, control unit, and the tubing is for the creation and the maintenance of the vacuum. The inclusion of the hyperspectral cameras is for the identification of soil composition and the selection of sample collection site. The reason for two different hyperspectral cameras is to have a wider spectral range. The NIR-640 has a spectral range of 900-1700nm and the XC2 has a spectral range of 400-1000nm. The outdoor field system is for the hyperspectral cameras to be used by the astronauts and includes a tripod with a laptop tray, a laptop preloaded with Spectronon software, a rotational scanning stage, and two calibration standards. The field system all comes in a travel case and will also house the camera. The usage of the travel case is for easier storage and better protection.

4.1.2 Scientific Equipment and Sample Collection Methods

1. Sample Collection Methods
    a. Comparison and use cases
    b. Decision Matrix
    c. Mechanisms, Specific Values
    d. Cubesats/Orbital Instruments
    e. Sample Collection and Storage
2. Mars Landers
    a. Zuhang Rover on Tianwan-1 Lander
        i. **Orbiter Instruments**
        Moderate Resolution Imaging Camera (MoRIC) Color photos, visible band, resolution 100m from 400km altitude

High Resolution Imaging Camera (HiRIC) Resolution of 2.5m from 256km altitude panchromatic mode, 10m in color mode.
Mars Orbiter Magnetometer (MOMAG) maps magnetic fields
Mars Mineralogical Spectrometer (MMS) utilizes visible and near infrared imaging spectrometer to analyze surface composition, subsurface structure
Mars Orbiter Scientific Investigation Radar (MOSIR) explores water-ice by means of dual-polarization echo characteristics of RADAR
Mars Ion and Neutral Particle Analyzer (MINPA) measures flux of ions in space, distinguishes main ions and obtains physical parameters such as density, velocity, temperature
Mars Energetic Particle Analyzer (MEPA) obtains energy spectrum, flux and elemental composition of energy electrons, protons, alpha particles and ions

ii. **Rover Instruments**

Mars Rover Penetrating Radar (RoPeR) Ground-penetrating radar, two frequencies to image 100m below surface.
Mars Rover Megnetometer (RoMAG) obtains fine-scale structures of crustal magnetic fields while moving over the surface
Mars Climate Station (MCS) (or Mars Meterological Measurement Instrument, MMMI) measures temp, pres, wind velocity and direction, incorporated microphone.
Mars Surface Compound Detector (MarSCoDe) laser-induced breakdown spectroscopy (LIBS) and infrared spectroscopy
Multispectral Camera (MSCam) Combined with MarSCoDe, investigates mineral components and searches for historical environmental conditions
Navigation and Topography Cameras (NaTeCam) 2048x2048 resolution, constructs topography maps, measures slope, undulation, roughness, performs comprehensive analysis on geological structures of surface parameters

    b. Perseverance Rover
        i. Cached sample tubes, 1kg samples of rock and atmosphere launched to be picked up later from low Martian Orbit. 4 tubes launched, 43 total 'witness'.
        Sample Drill- abrases rock, cleans dust away by on-board high-pressure nitrogen, drills out core into sample container. Container

needs to be able to survive terminal velocity impacts to avoid contamination.

ii. Seven Primary Payload Instruments

iii. Nineteen Cameras

iv. Two microphones

v. One deployable mini-helicopter (ingenuity)

vi. Terrain Relative Navigation (TRN) compares surface images during decent with reference maps to make adjustments and identify safe landing site.

c. Mars 5M (Russian, 1980, cancelled)

    i. 2 Proton Rockets sent 8500kg to mars, land, collect samples, and separate with 2000kg stage returning to Mars orbit to rendezvous with return spacecraft delivered by another Proton. Samples sterilized by heat, land on Earth without parachutes and located via radioactive beacon.

d. InSight

    i. Robotic lander for interior study.

    ii. Seismometer (SEIS) measures seismic activity

    iii. Heat Flow and Physical Properties Package (HP^3) Radiometer and heat probe, burrows 5m below surface trailing heat sensors every 10cm

    iv. Rotation and Interior Structure Experiment (RISE) X-band radio to measure rotation, accurate to 2cm, calculates size and density of core and mantle

    v. Temperature and Winds for Insight (TWINS) monitors weather

    vi. Laser RetroReflector for Insight (LaRRI)- retroreflector enables passive laser range-finding by orbiters. Used to map geophysical network

    vii. Instrument Deployment Arm (IDA) 1.8m robotic arm that deploys instruments to surface. 4DOF motorized manipulator, constructed from carbon-fiber composite tubes, with scoop, wax actuated grappling claw, IDC camera.

    viii. Instrument Deployment Camera (IDC) color camera, 1024x1024 resolution, 45°

    ix. Instrument Context Camera (ICC) Same but wide-angle, 120 degree panorama

    x. Star-tracker for Navigation

e. Schiaparelli EDM Lander

    i. DREAMS (Dust Characterization, Risk Assessment, and Environmental Analyzer on the Martian Surface) suite for wind detection, humidity, pressure, temp, solar irradiance (transparency of atmosphere), atmospheric electricity detector, Descent Camera, Combined Aerothermal Sensor Package

f. Phoenix Lander

    i. Thermal and Evolved Gas Analyzer (TEGA) High temp furnace with mass spectrometer, bakes samples of dust and measures vapors.

ii. Miscroscopy, Electrochemistry, Conductivity Analyzer (MECA)
iii. Optical and Atomic Force Microscope. 2mm x 2mm and 0.1micrometer res.
iv. Wet Chemistry Laboratory (WCL) Scoops soil, adds water, measures dissolved ions leaching from soil.

g. Opportunity/Spirit Rover
    i. Cameras, Spectrometers, Microscope Imager
    ii. Rock Abrasion Tool (RAT) exposes fresh material, grind and brushing installation to gain access to interior of rocks to provide other instruments with a smooth, clean surface to study

h. Phobos 1 and 2 Landers, 1988
    i. PROP-F 'hopping lander' x-ray fluorescence spectrometer, ferroprobe magnetometer, kappameter magnetic permeability/suspectibility sensor, gravimeter, temperature sensors, BISIN conductometer/tiltmeter, mechanical sensors (penetrometer, UIU accelerometer, sensors on hopping mechanism)
    ii. DAS (long-lived autonomous station) lander- TV camera, Alpha-Proton-X-Ray Spectrometer, seismometer, infrared spectrometer/radiometer, thermal imagine camera, magnetometers, x-ray telescope, radiation detectors, radar and laser altimeters, 'grunt' imaging radar

i. Martian Moons Exploration (MMX), robotic space probe
    i. Will collect samples from Phobos by landing once or twice and gathering sand particles using a simple pneumatic system, up to 10g. Will take off, make several flybys of the smaller moon Deimos before sending return module back to Earth.
    ii. Three modules: Propulsion Module (1800kg) Exploration Module (150kg) Return Module (1050kg) Deimos and Phobos mass are too small to capture a satellite so quasi-satellite orbits are performed
    iii. TENGOO - TElescopic Nadir imager for GeOmOrphology, a narrow field camera for detailed terrain study
    iv. OROCHI - Optical RadiOmeter composed of CHromatic Imagers, a wild field visible light camera
    v. LIDAR - Light Detection and Ranging, uses a laser to reflect light from the moon's surface, to study surface altitude and albedo
    vi. MIRS - MMX InfraRed Spectrometer, a near-infrared observation device for characterizing the minerals that make up the moons of Mars. Developed in partnership with CNES, France
    vii. MEGANE - (MEGANE means "eyeglasses" in Japanese) Mars-moon Exploration with GAmma rays and NEutrons, a gamma-ray and neutron spectrometer developed in partnership with NASA
    viii. CMDM - Circum-Martian Dust Monitor, a dust counting device for characterizing the environment around the Martian moons
    ix. MSA - Mass Spectrum Analyzer, an instrument to study the ion environment around Mars
    x. Super-Hi-Vision Camera- 4k and 8k camera resolution.

xi. Gravity Gradiometer (GGM)
xii. Laser-Induced Breakdown Spectroscope (LIBS)
xiii. Mission Survival Module (MSM)
xiv. Coring Sampler (C-SMP) to gain regolith deeper than 2cm
xv. Pneumatic Sampler (P-SMP) air gun puffs pressurized gas, collects 10g soil

j. Fobos-Grunt Russian Lander, 2011
   i. Return stage launched by springs so as not to damage lander components, accelerated to 35km/h to escape Phobos gravity

3. Moon Landers
   a. Apollo 17
      i. LRV to carry Traverse Gravimeter and Surface Electrical Properties Experiment
      ii. Biological Cosmic Ray Experiment
      iii. Scientific Instrument Module (SIM) bay from orbit- lunar sounder for geological model to depth of 1.3km, infrared scanning radiometer for temperature map of surface, far-ultraviolet spectrometer for lunar atmosphere composition, density, laser altimeter
      iv. ALSEP (Apollo Lunar Surface Experiments Package) full suite of seismic, magnetic, ion detection, solar wind and lunar heat instruments
   b. Chandrayaan 2 (india)
      i. Orbiter
      ii. Chandrayaan-2 orbiter in clean-room being integrated with payloads
      iii. Payloads on the orbiter are
      iv. Chandrayaan-2 Large Area Soft X-ray Spectrometer (CLASS) from the ISRO Satellite Centre (ISAC), which makes use of X-ray fluorescence spectra to determine the elemental composition of the lunar surface
      v. Solar X-ray monitor (XSM) from Physical Research Laboratory (PRL), Ahmedabad, primarily supports CLASS instrument by providing solar X-ray spectra and intensity measurements as input to it. Additionally these measurements will help in studying various high-energy processes occurring in the solar corona.
      vi. Dual Frequency L-band and S-band Synthetic Aperture Radar (DFSAR) from the Space Applications Centre (SAC) for probing the first few metres of the lunar surface for the presence of different constituents. DFSAR was expected to provide further evidence confirming the presence of water ice, and its distribution below the shadowed regions of the Moon. It has lunar surface penetration depth of 5 m (16 ft) (L-band).
      vii. Imaging IR Spectrometer (IIRS) from the SAC for mapping of lunar surface over a wide wavelength range for the study of minerals, water molecules and hydroxyl present. It featured an extended spectral range (0.8 μm to 5 μm), an improvement over previous lunar missions whose payloads worked up to 3 μm.

viii. Chandrayaan-2 Atmospheric Compositional Explorer 2 (ChACE-2) Quadrupole Mass Analyzer from Space Physics Laboratory (SPL) to carry out a detailed study of the lunar exosphere
ix. Terrain Mapping Camera-2 (TMC-2) from SAC for preparing a three-dimensional map essential for studying the lunar mineralogy and geology
x. Radio Anatomy of Moon Bound Hypersensitive Ionosphere and Atmosphere – Dual Frequency Radio Science experiment (RAMBHA-DFRS) by SPL for the studying electron density in the lunar ionosphere
xi. Orbiter High Resolution Camera (OHRC) by SAC for scouting a hazard-free spot prior to landing. Used to help prepare high-resolution topographic maps and digital elevation models of the lunar surface. OHRC had a spatial resolution of 0.32 m (1 ft 1 in) from 100 km (62 mi) polar orbit, which was the best resolution among any lunar orbiter mission to date.
xii. Vikram lander
xiii. The payloads on the Vikram lander were:
xiv. Instrument for Lunar Seismic Activity (ILSA) MEMS based seismometer by LEOS for studying Moon-quakes near the landing site
xv. Chandra's Surface Thermo-physical Experiment (ChaSTE) thermal probe by SPL, Vikram Sarabhai Space Centre (VSSC) for estimating the thermal properties of the lunar surface
xvi. RAMBHA-LP Langmuir probe by SPL, VSSC for measuring the density and variation of lunar surface plasma
xvii. A laser retroreflector array (LRA) by the Goddard Space Flight Center for taking precise measurements of distance between the reflector on the lunar surface and satellites in lunar orbit.
The microreflector weighed about 22 g (0.78 oz) and cannot be used for taking observations from Earth-based lunar laser stations.
xviii. Pragyan rover
xix. Pragyan rover carried two instruments to determine the abundance of elements near the landing site:
xx. Laser induced Breakdown Spectroscope (LIBS) from the laboratory for Electro Optic Systems (LEOS), Bangalore
xxi. Alpha Particle Induced X-ray Spectroscope (APXS) from PRL, Ahmedabad

c. Chang’e (China)
i. Landing Camera
ii. Panoramic Camera
iii. Lunar Mineralogical Spectrometer
iv. Lunar Regolith Penetrating Radar
v. Subsurface sample drill

vi.Surface scoop device

4. Mission Objectives
    a. Mars Energetic Particle Analyzer/Radiation Detector/Alpha Magnetic Spectrometer
    b. Chemical Analysis on Soil
    c. Look for Biomolecules and Biosignatures
    d. Study Geological Structure
    e. Study Characteristics of surface and underground layers
    f. Study Composition and Type of rocks- Spectrometer and Multispectral Cameras
    g. Study Atmospheres- Particle Detectors
    h. Study Internal Structure, Magnetic Fields, History of Geological Evolution, Distribution of Mass, Gravitational Field- Magnetometers and RADAR
    i. Build surface maps- High Resolution Cameras

5 Spacecraft Subsystems

5.1 Propulsion

5.1.1 Primary Propulsion System

5.1.1.1 Fuel Selection

A key constraint in this project, and most other space missions, is mass. For this reason, an efficient propellant is naturally desirable as the amount of dV that can be extracted from every kilogram of propellant is directly proportional to specific impulse per the rocket equation.

Equation 1: The rocket equation

$$\Delta v = \ln\left(\frac{m_f}{m_e}\right) * g_0 * I_{sp}$$

Based on propellant efficiency, an obvious choice is liquid hydrogen and liquid oxygen. This bipropellant system was what drove the Space Shuttle’s RS-25 engines which are still some of the greatest rocket engines ever made. However, there is a critical drawback. Hydrogen is the least dense element and must be condensed to its liquid form in order to have a useful amount stored in a fuel tank. This also requires that it be cryogenically cooled to an -253˚C (-423˚F), which takes a substantial amount of energy. For launch vehicles supported by umbilical harness systems, this is not much of a problem as freshly chilled propellant and oxidizer can be continuously cycled up to the moment of liftoff. However, a spacecraft which must remain in space for several years, directly exposed to solar radiation, keeping the hydrogen cryogenically cooled becomes a monumental task.

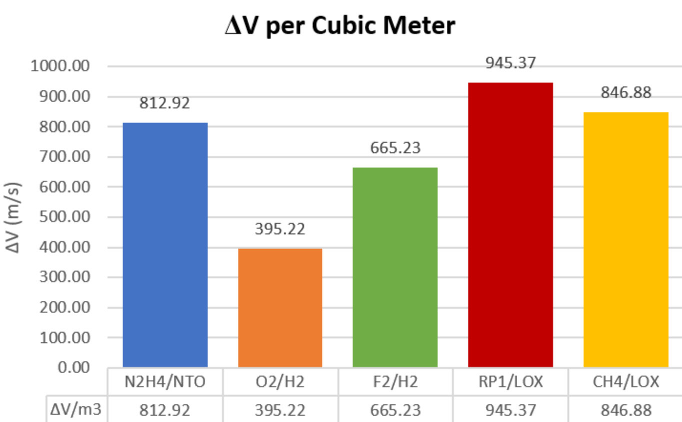


Figure 6: Propellants compared by volume

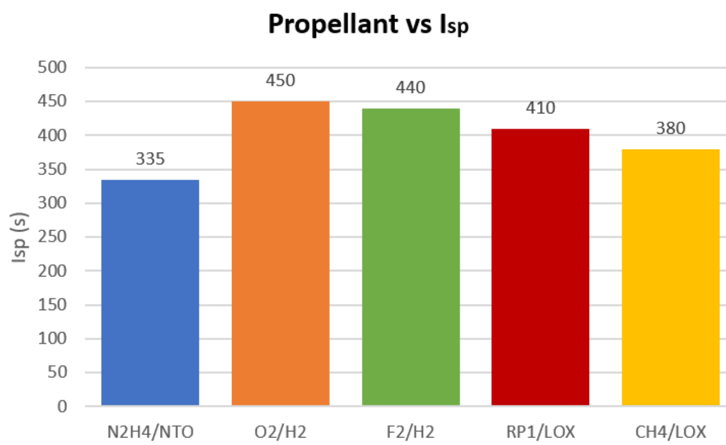


Figure 7: Propellants compared by I_{sp}

This is the point where hypergolic propellant systems, such as hydrazine and nitrogen tetroxide, emerge as attractive choices. Hypergolic systems require no additional starting mechanism as the fuel and oxidizer combust immediately upon contact. Many hypergolic fuels are also liquid at room temperature, negating the need for cryogenic cooling for both the fuel and the oxidizer.

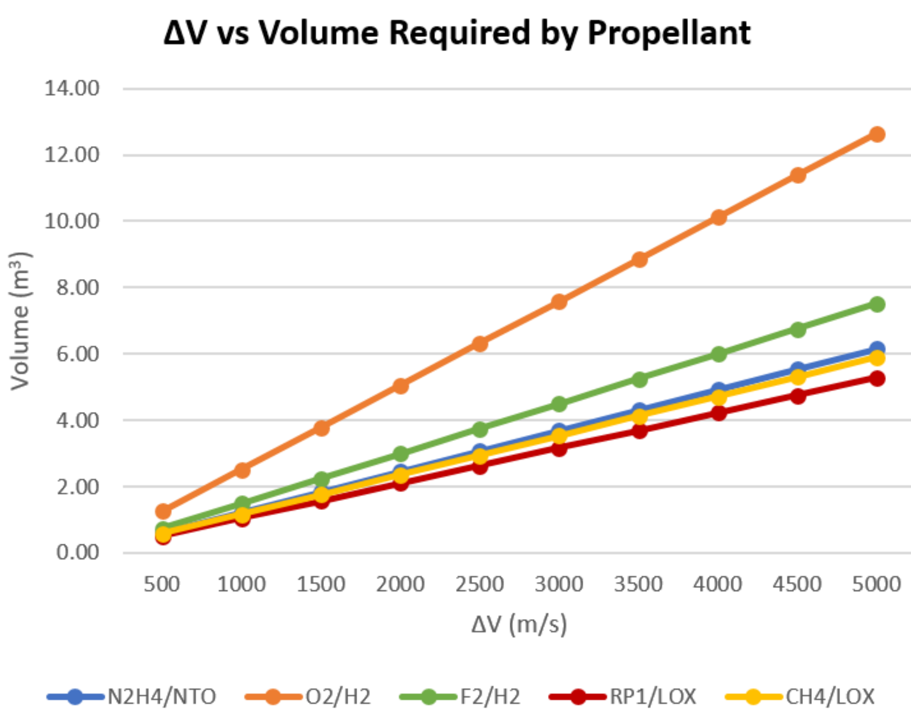


Figure 8: Propellant volume required by dV

5.1.1.2 Engine Selection

With a hydrazine/nitrogen tetroxide fuel-oxidizer system selected for the primary propulsion system, a corresponding engine was required. Several historical and currently operational hydrazine-based vacuum engines were compared by availability, mass, and engine efficiency. Certain engines, such as the TR-201, feature desirable properties and served as good points of comparison but were disqualified from selection due to their real-world availability, which was most often limited by the engine being retired or by an engine being produced by a country which would make procurement of the engine impossible.

Table 2: Hydrazine-based vacuum engine comparison

Engine	I_{sp} (s)	Mass (kg)	Length (m)	Diameter (m)	Status	Thrust (N)	Insertion Burn Time (s)
AJ10	319	100	1.96	0.84	Operational	43.7E+3	202.05
TR-201	301	113	2.27	1.38	Retired	41.9E+3	206.28
Aestus II	340	138	2.29	1.31	Operational	55.4E+3	162.97
S5.80	302	310	1.2	2.1	Operational	3.0E+3	2933.60
RD-253	316	1080	3	1.5	Operational	1.6E+6	5.40
RD-270	322	3370	4.9	3.4	Retired	6.7E+6	1.32

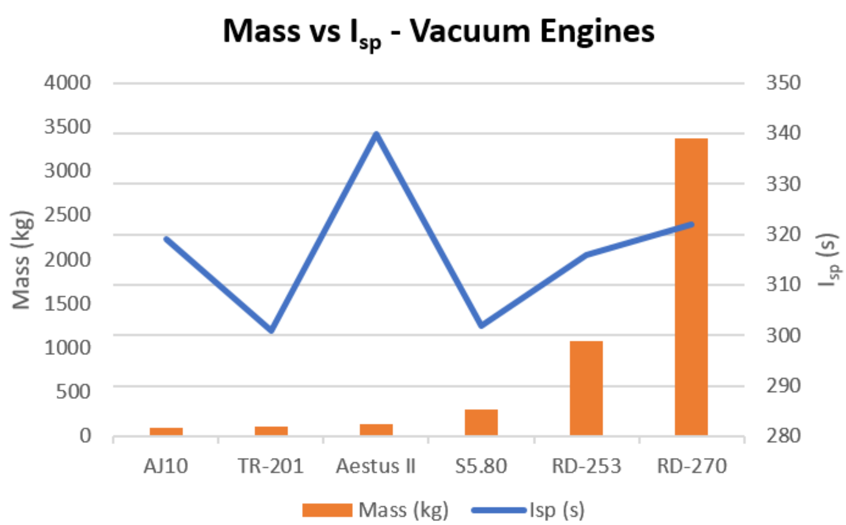


Figure 9: Existing vacuum engines by mass and I_{sp}

Table 2 and Figure 9 offer a comparison of engines which were considered for this mission, with the most promising among them subsequently compared in Figure 10. From these, the clear choice for this mission is the Aestus II which is currently being developed by the German company Astrium in association with the European Space Agency for use as an upper stage engine for future variants of the Arianne 5 rocket family. A close runner-up is the Aerojet Rocketdyne AJ10 engine, most notably used for the Space Shuttle's Orbital Maneuvering

System (OMS) and for the upcoming NASA Orion spacecraft. A key advantage the AJ10 has over the Aestus II is that it is currently operational and has an extensive, proven history of performance compared to the Aestus II, which is still being developed.

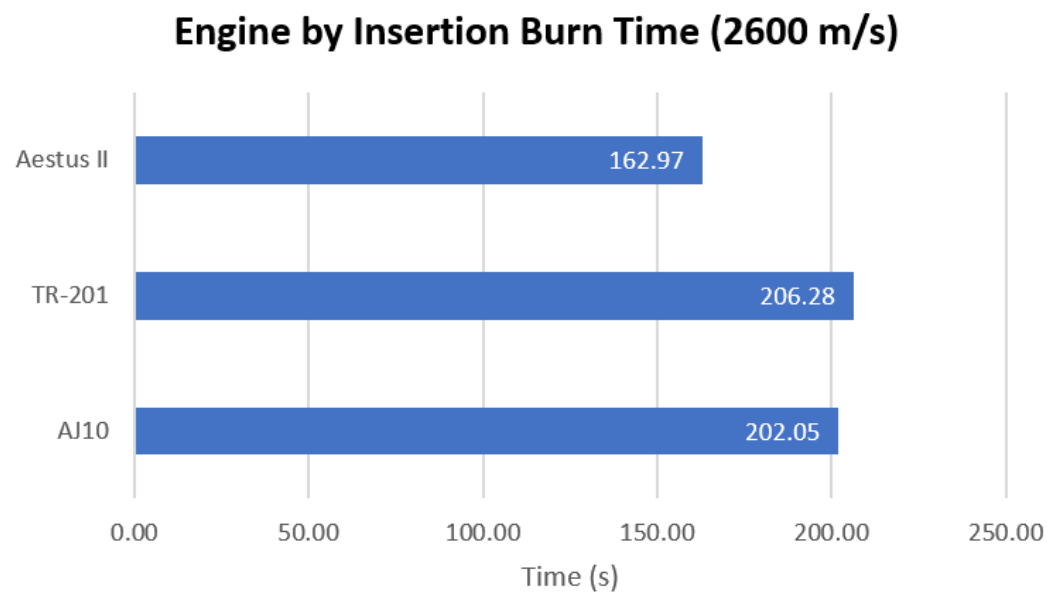


Figure 10:The Aestus II displays greater thrust and efficiency

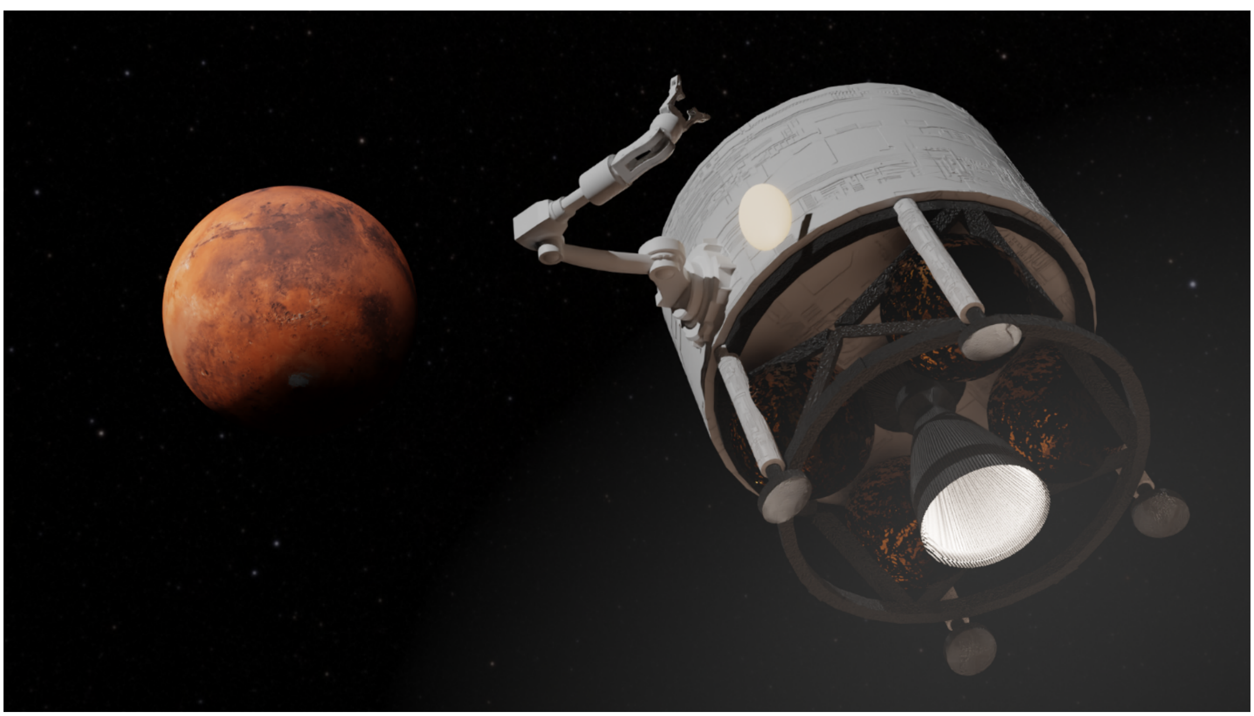


Figure 11: Render of an early iteration of LUPA approaching Mars with her Aestus II engine burning

5.1.2 Reaction Control System

5.1.2.1 Fuel Selection

A supporting factor for the selection of hydrazine for the primary propulsion system is its ability to also be utilized as a reaction control propellant. Many spacecraft utilize a series of small thrusters which sacrifice efficiency for precise control. These “monopropellant” systems required only a pressurized fuel to be passed over a catalyst to produce thrust.

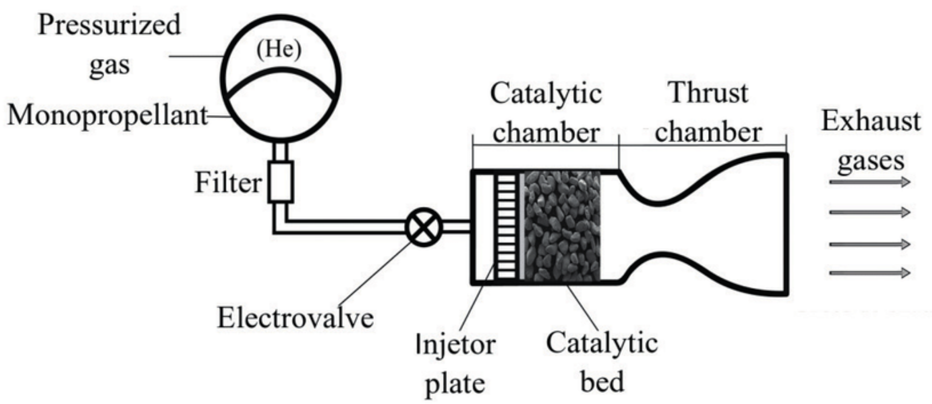


Figure 12: Diagram of a monopropellant thruster

5.1.2.2 Thruster Selection

Many real-world vehicles already use hydrazine-based RCS thrusters, so an abundance of choices are available to be applied to LUPA. For this reason, the Aerojet Rocketdyne MR-104 thruster was selected. This thruster is currently being used by the in-development NASA Orion manned spacecraft which provides the benefit of unifying the architecture of LUPA with that already in development at NASA.

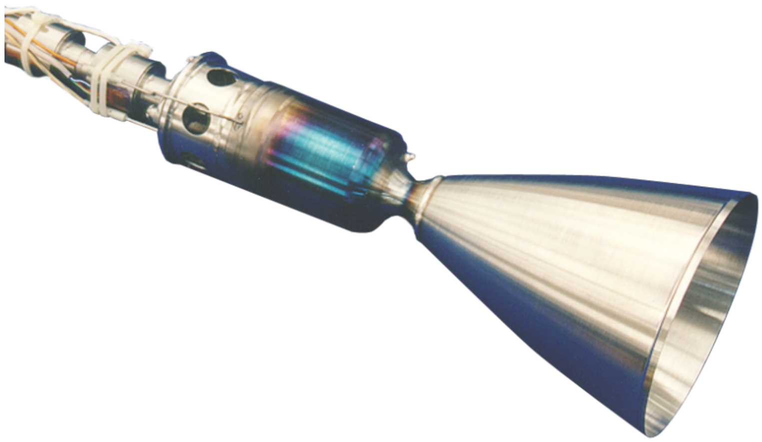
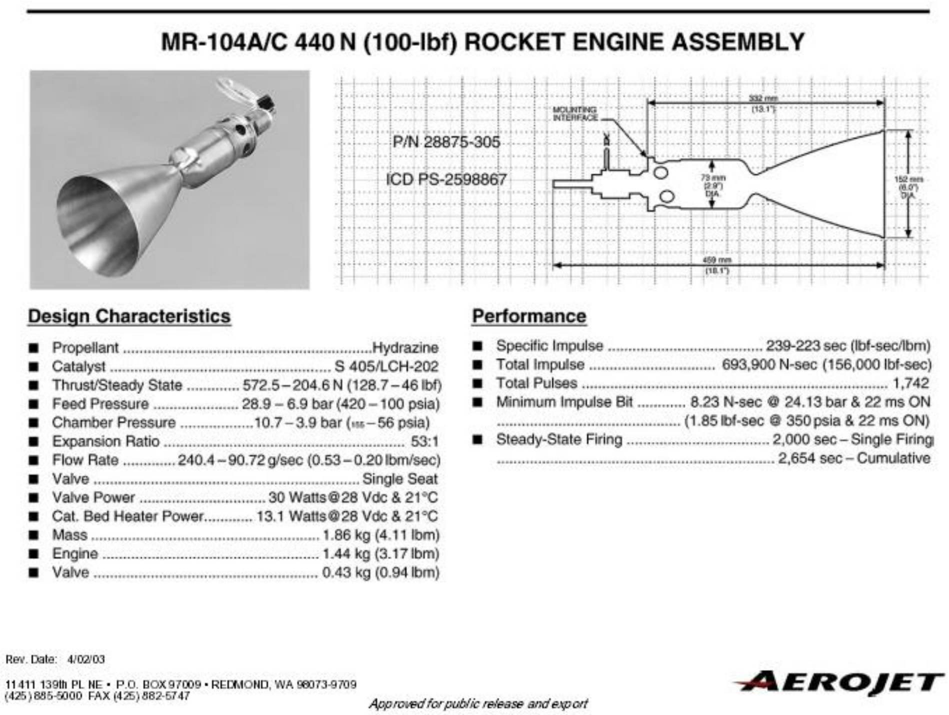


Figure 13: The MR-104 thruster



MR-104A/C 440 N (100-lbf) ROCKET ENGINE ASSEMBLY

P/N 28875-305

ICD PS-2598867

Design Characteristics

- Propellant .... Hydrazine
- Catalyst .... S 405/LCH-202
- Thrust/Steady State .... 572.5 – 204.6 N (128.7 – 46 lbf)
- Feed Pressure .... 28.9 – 6.9 bar (420 – 100 psia)
- Chamber Pressure .... 10.7 – 3.9 bar (155 – 56 psia)
- Expansion Ratio 53:1
- Flow Rate .... 240.4 – 90.72 g/sec (0.53 – 0.20 lbm/sec)
- Valve .... Single Seat
- Valve Power .... 30 Watts @ 28 Vdc & 21°C
- Cat. Bed Heater Power........ 13.1 Watts @ 28 Vdc & 21°C
- Mass .... 1.86 kg (4.11 lbm)
- Engine .... 1.44 kg (3.17 lbm)
- Valve .... 0.43 kg (0.94 lbm)

Performance

- Specific Impulse .... 239-223 sec (lbf-sec/lbm)
- Total Impulse .... 693,900 N-sec (156,000 lbf-sec)
- Total Pulses 1,742
- Minimum Impulse Bit .... 8.23 N-sec @ 24.13 bar & 22 ms ON .... (1.85 lbf-sec @ 350 psia & 22 ms ON)
- Steady-State Firing .... 2,000 sec – Single Firing .... 2,654 sec – Cumulative

Rev. Date: 4/02/03

11411 139th PL NE • P.O. BOX 97009 • REDMOND, WA 98073-9709
(425) 885-5000 FAX (425) 882-5747

Approved for public release and export

AEROJET

Figure 14: MR-104 thruster datasheet

5.2 ECLSS

5.2.1 Foodstuffs

The matter of nutrition is spaceflight is a complicated subject as according to NASA's Space Food Systems, "The Space Food Systems Laboratory produces freeze-dried food and packages commercially available beverage powder, cookies, candy, and other dried goods that astronauts select for their menus." [Douglas, Läte, and Wu] NASA also repackages commercially available products. [Savage] These two facts make accurate estimation of the mass or volume of any meals used on the mission impossible until after construction of the spacecraft due to all meals being astronaut selected.

To combat this complication Meals, Ready-to-Eat (referred to as MREs) were used to provide a rough estimation of mass and volume. To reduce the mass and volume of a typical MRE, the MREs were field stripped according to the method used by a wildland firefighter. [Kennard] After field stripping an MRE's mass is 0.83kg while it's dimension are a 152.4mm x 50.8mm x 181mm box.

For a fifteen day mission, three meals a day, and two astronauts, the minimum number of meals needed is ninety. To provide a safety factor of nearly two, one hundred sixty-eight meals will be used. The mass of one hundred sixty-eight MREs comes to 139.440kg. When arranged in a group six wide, seven tall, and four deep, their dimensions are 914.4mm x 355.6mm x 723.9mm.

5.3 Power Distribution

5.3.1 Power Generation

LUPA is similar in overall size and crew-habitation capability of the SpaceX Dragon 2 spacecraft, whose solar arrays provide for up to 2 kilowatts of electricity. The Dragon 2 features fixed solar cells along its aft "trunk" section, differing from the deployable solar panels of the original Dragon vehicle. The cells on the Dragon 2 are more efficient than those used on the Dragon's deployable arrays and serve to reduce complexity by requiring no moving parts. Since LUPA is scheduled to arrive at Mars 4 years before the crew to pilot her, this reduced complexity is ideal. Preliminary estimates indicate a necessary panel area of 12.42 square meters, which is approximately the projected area of the most current iterations of LUPA.

Similar estimates based on NASA's "Roll Out" solar arrays currently aboard the ISS yield a required area of only 4 square meters due to their greatly increased efficiency. In the final design report, a more detailed trade study comparing the mass savings versus complexity of these two systems will be used to justify the final configuration of LUPA's power delivery system.

Figure 15: The Dragon 2's fixed solar array

LUPA will also feature a standard set of batteries for energy storage based on those used in vehicles such as Soyuz or Dragon amounting to an additional 200 kilograms.

5.4 Thermal Management

Based on the 3 kilowatt maximum power requirement and an estimated efficiency of 90%, LUPA is estimated to produce, at maximum power usage, 300 watts of heat which needs to be radiated away from the spacecraft in order to protect the crew and vital components. To radiate this amount of heat, approximately half a square meter of radiating area will be required.

6 Structure

Structural design requirements are defined below [Zito]

1) Design Loads
    a) $58.86 m/s^2$ axial acceleration

b) 16.905 psi of pressure

2) Ultimate Loads
   a) 88.29m/s^2 axial acceleration
   b) 29.4 psi of pressure

The normal operational pressure load comes from NASA using 14.7psi of pressure for the atmosphere of the space shuttle and the International Space Station while the maximum operational axial load is 58.86m/s^2 of acceleration from SpaceX's Falcon Heavy. [Falcon User's Guide 17-18]

6.1 Pressure Vessel, Hull, and Debris Shield

6.1.1 Pressure Vessel

The pressure vessel is the pressurized portion of LUPA that will contain the atmosphere required for the astronaut to survive. The pressure vessel is made of 3mm thick aluminum 2219-T62. From top to bottom the pressure vessel is 5663.3547m tall and at its max it is 4000mm is diameter. The top and bottom of the pressure vessel feature a conical section that reduces in diameter to 1500mm during the proceeding 300mm. The inclusion of the conical section occurred during version six to better distribute the stress across the pressure vessel as well as reduce the amount of stress at concentration points. The top of the pressure vessel features an 800mm diameter hole [Gerstenmaier 3-5] out into the pressure vessel. This hole is where the docking hatch and transfer tunnel will be placed. Due to its size and thickness the pressure vessel is the heaviest piece of the spacecraft at 756.1951 kilograms.

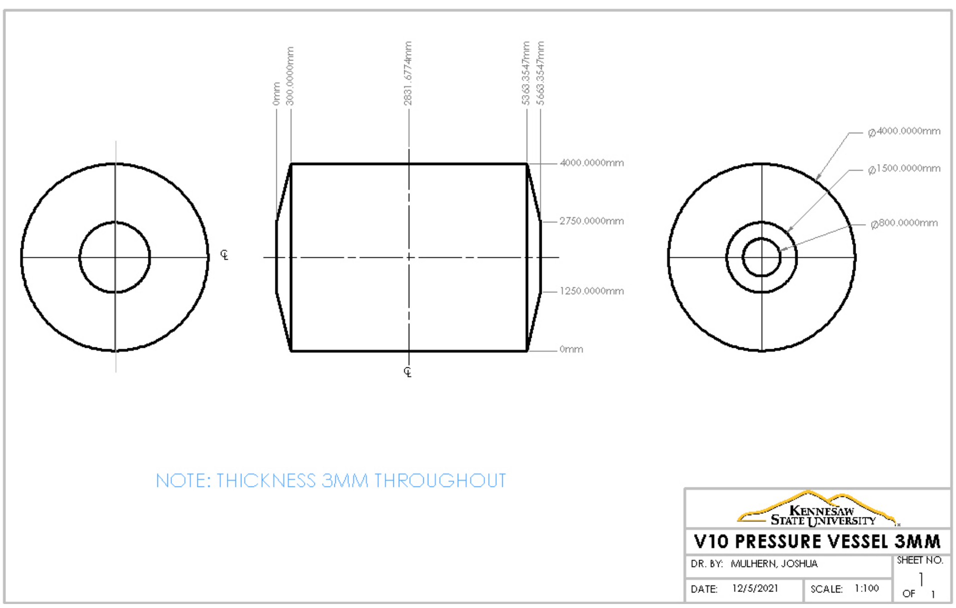


Figure 16: Version ten pressure vessel with 3mm thick walls

6.1.2 Hull

The hull is constructed of 0.41mm thick aluminum 2219-T62 It is placed between the pressure vessel and the debris shied and features the same shape as the pressure vessel. Also similar to the pressure vessel the top side has an 800mm diameter hole cut out of it. The hull is 5863.4367mm tall and 4200.082mm in diameter. When placed around the pressure vessel, all surfaces of the hull are offset 100mm from the pressure vessel. Due to its thinness the hull only weight 11.323kg.

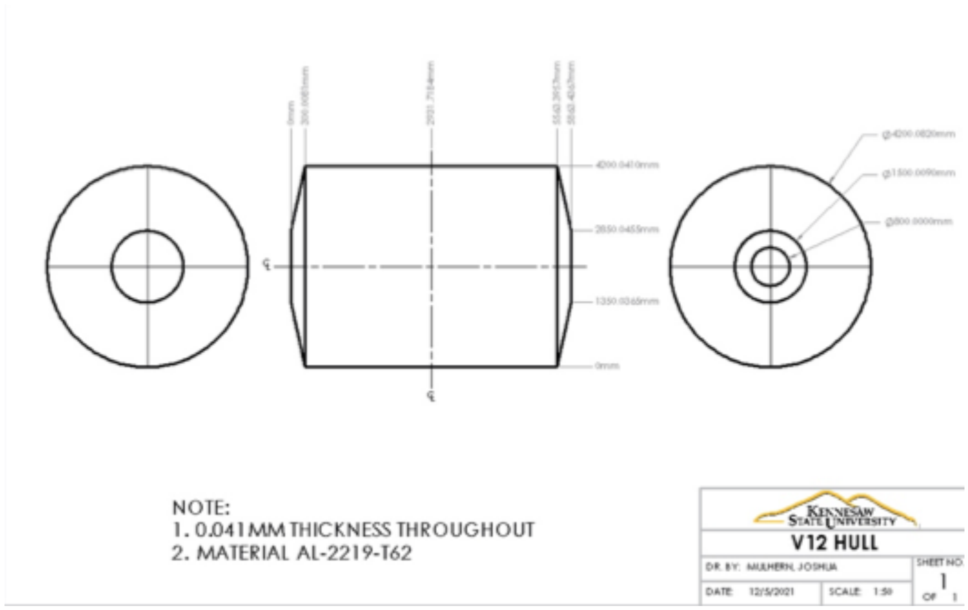


Figure 17: Version twelve hull drawing

6.1.3 Debris Shield

The debris shield is a vital part of LUPA's design as it protects it from micrometeorite strikes. As no information on the number of micrometeorites around mars or its moons Phobos and Deimos could be found, the debris shield was integrated. When combined with the hull and the pressure vessel, the three form a triple wall Whipple shield. The distances and materials of the pressure vessel, hull, and debris shield are according to NASA/TM-2009-214789. [Ryan and Christiansen 63] The only difference is that the rear wall is replace with a monolithic shield that is 3mm thick. This should decrease the penetration rate. It is constructed of 0.41 thick aluminum 2219-T62. It is 5935.815mm tall, 4272.46mm in diameter and weighs 116.101kg. When placed around the spacecraft, the debris shield is offset 35mm from the hull.

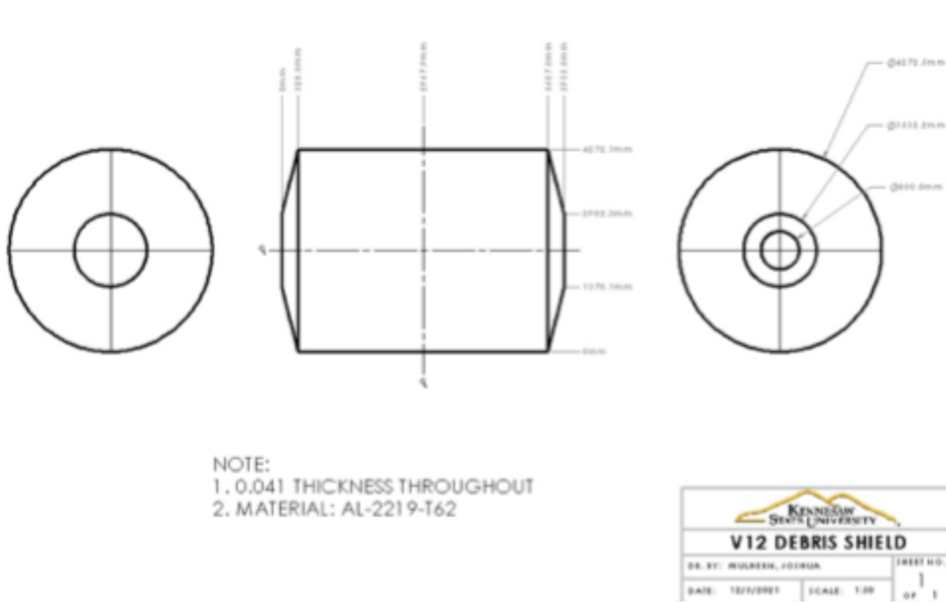


Figure 18: Version twelve debris shield drawing

6.1.4 Arrangement

When the debris shield, the hull, and pressure vessel, are combined the three form a triple wall Whipple shield. The distances and materials of the pressure vessel, hull, and debris shield are according to NASA/TM-2009-214789. [Ryan and Christiansen 22] The only difference is that the rear wall is replace with a monolithic shield that is 3mm thick. This should decrease the penetration rate.

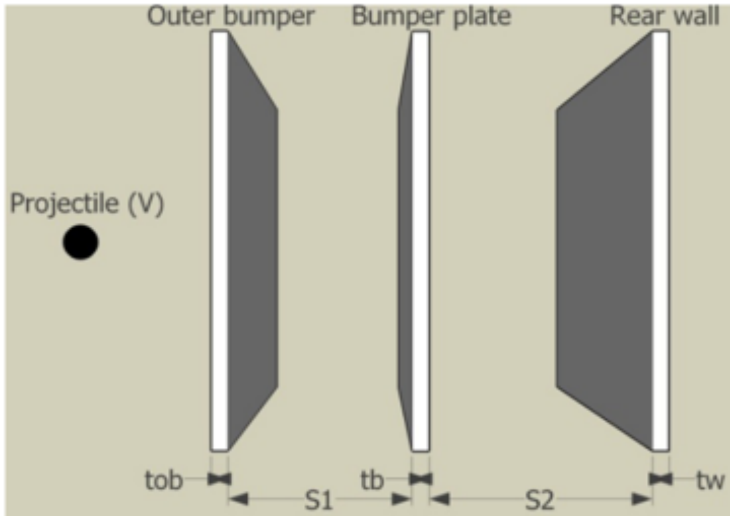


Figure 19: Triple Whipple shield cross-section

6.1.5 Landing Legs

Due to the microgravity environment of the moons of Mars, traditional landing legs are unnecessary. Instead, thin, probe-like aluminum rods akin to flexible tent poles used in modern camping equipment can be utilized. The main purpose of these far-less massive rods will not be to support the weight of the vehicle on the surface, as traditional landing legs would, but rather to provide a tactile feedback system when approaching the surface. These rods will also serve to prevent unwanted rotation of the capsule while the crew conducts surface operations. The arrangement and dimensions of these legs will be defined fully in the subsequent final design review.

6.2 Frameworks

6.2.1 Stringer Frameworks and Standoff Distance Frameworks

Both the stringer and standoff distance frameworks are constructed from Aluminum 6061-T6 I-beams. The stringer framework is place between the pressure vessel and the hull while the standoff distance framework is place between the hull and the debris shield. Proportions of the I-beams was based off a four-inch I-beam from McMaster-Car and then were resized to be 100mm and 35mm tall.

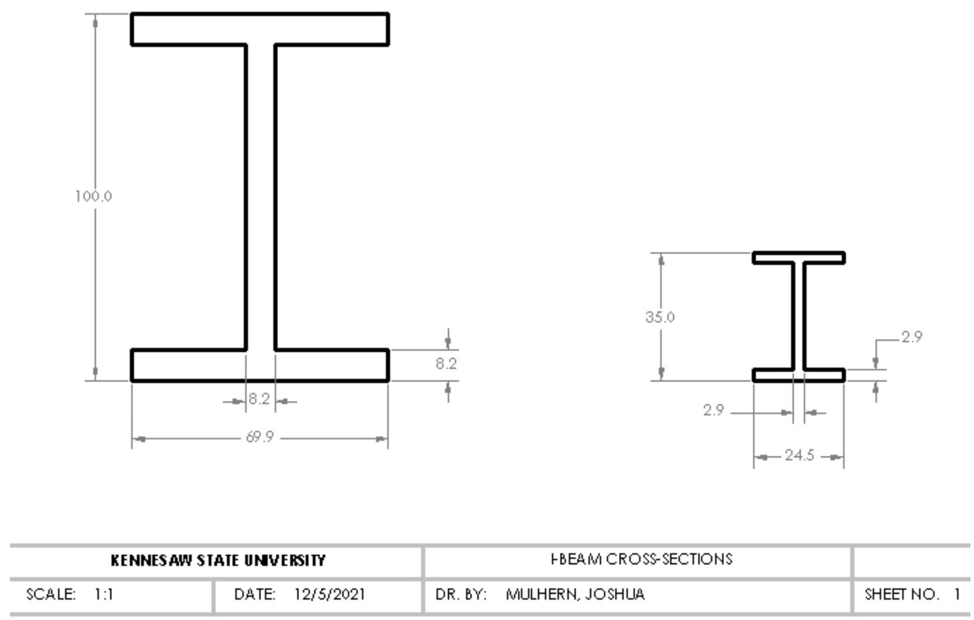


Figure 20: I-beam cross-section for stringer frameworks and standoff distance frameworks

6.2.2 Top and Bottom Frame

The top and bottom stringer frame consists of 2 circular hoops with five straight I-beam rotated radially around the center between the two hoops.

Table 3:Top and bottom frame masses

Part	Mass (kg)
Stringer Top Frame	40.708
Stringer Bottom Frame	50.140
Standoff Top Frame	5.260
Standoff Bottom Frame	6.448

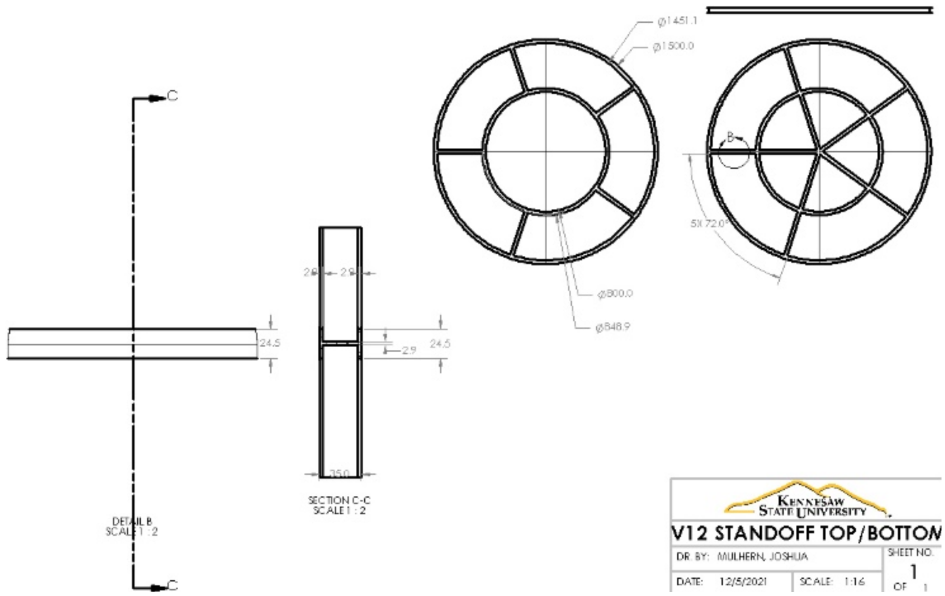


Figure 21: Version twelve top and bottom standoff distance frame

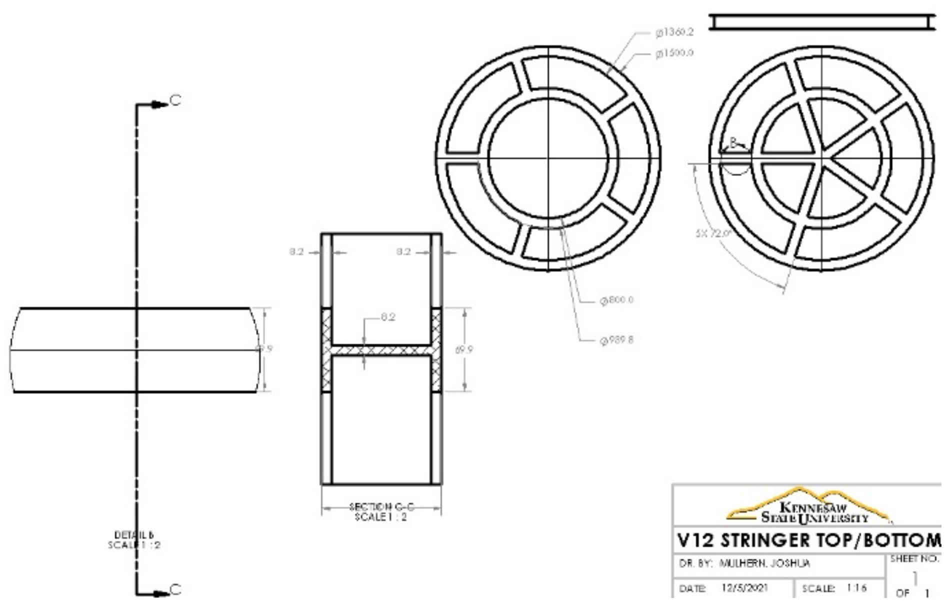


Figure 22: Version twelve top and bottom stringer frame

6.2.3 Diagonal Frame

The diagonal frame is the structural support that goes over the conical section of the pressure vessel and hull. The stringer diagonal frame consists of four hoops and ten straight I-Beams. The standoff diagonal stringer on the other hand only consists of two hoops and five straight I-beams.

For both the hoops are angled to make the conical section and reduce in diameter and they reach the top and bottom.

Table 4: Diagonal Frame Masses

Part	Mass (kg)
Stringer Diagonal Frame	219.734
Standoff Diagonal Frame	14.843

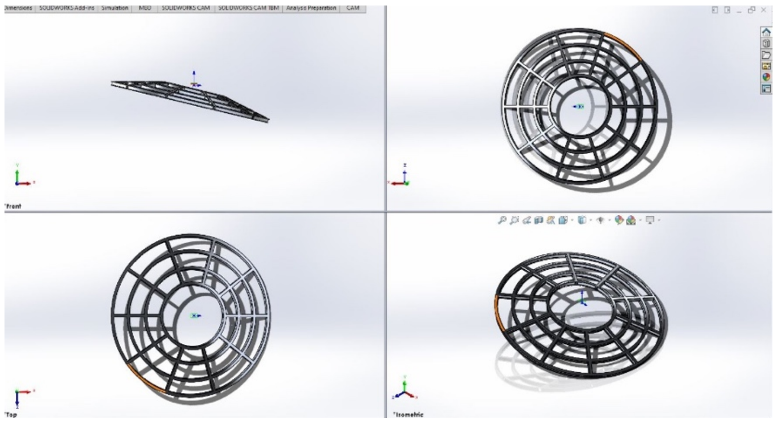


Figure 23: Version twelve stringer diagonal frame

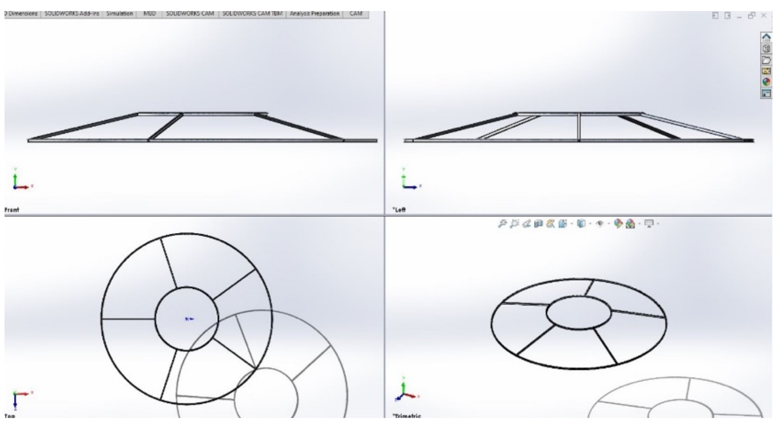


Figure 24: Version twelve standoff diagonal frame

6.2.4 Cage Frame

Both of the cage frame provide structure to the cylindrical section of the spacecraft. The stringer cage frame and the standoff cage frame both consist of three hoops and ten vertical I-beams. The three hoops are placed the beginning of both conical section and one in the middle of the spacecraft while the vertical beams are space evenly around the circumference of the spacecraft.

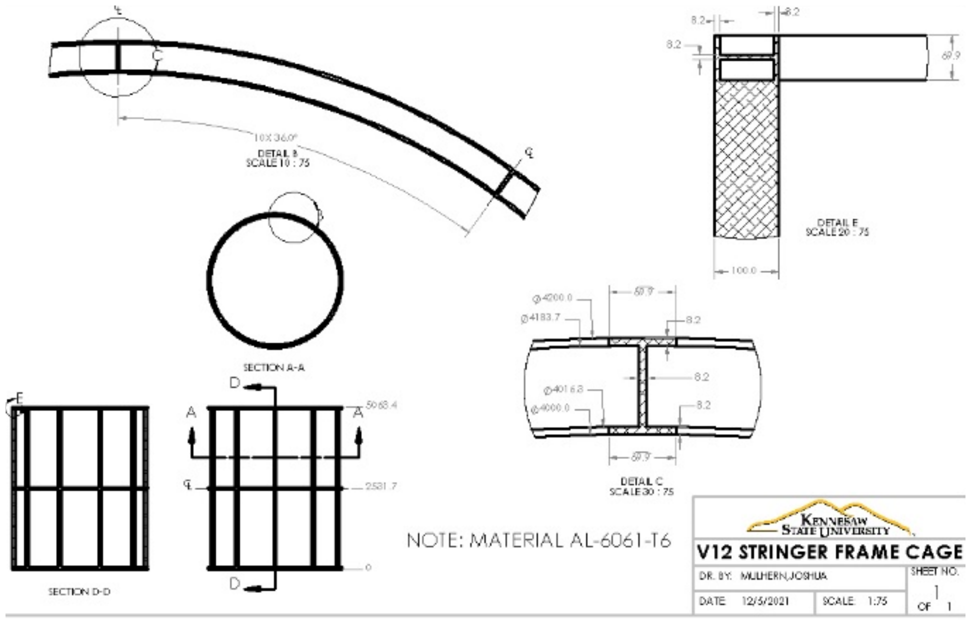


Figure 25: Version twelve stringer frame cage

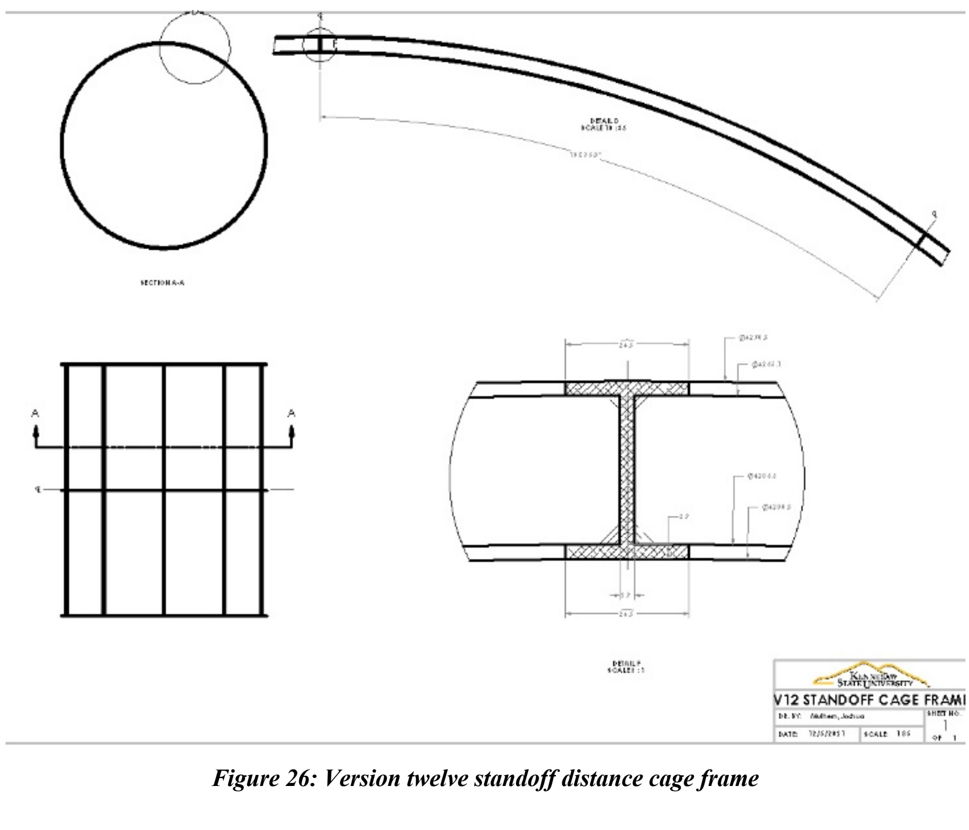


Figure 26: Version twelve standoff distance cage frame

7 Internal Layout

7.1 Cockpit

The cockpit is mounted to the side of the spacecraft and it functions as a dividing wall between the hatch and the rest of the spacecraft. Similar to Crew Dragon the cockpit is recessed back from the front of the spacecraft to allow for adequate space to use the docking hatch. [Chriara] On LUPA the front of the cockpit has 914.4mm between it and the docking hatch. The cockpit is mounted to the side of the spacecraft by way of a 304 thick L with 1mm wall. The reason for the overside thickness is to allow adequate room until it can be determined exactly how much space is need for wiring to the flight controls.

The layout of the cockpit itself is similar to the space shuttle. There are two flight chairs from the space shuttle [Crew Compartment] on either side of the cockpit with a dividing center console between the chair. Infront of the chairs are the flight controls and avionics required to pilot the spacecraft. Located just above eye level are the windows need for visual identification.

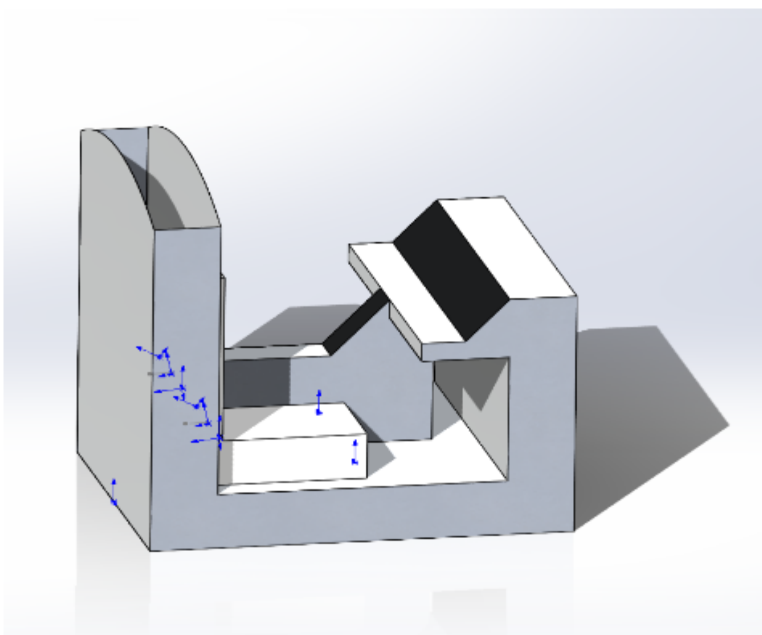


Figure 27: Cockpit Isometric View

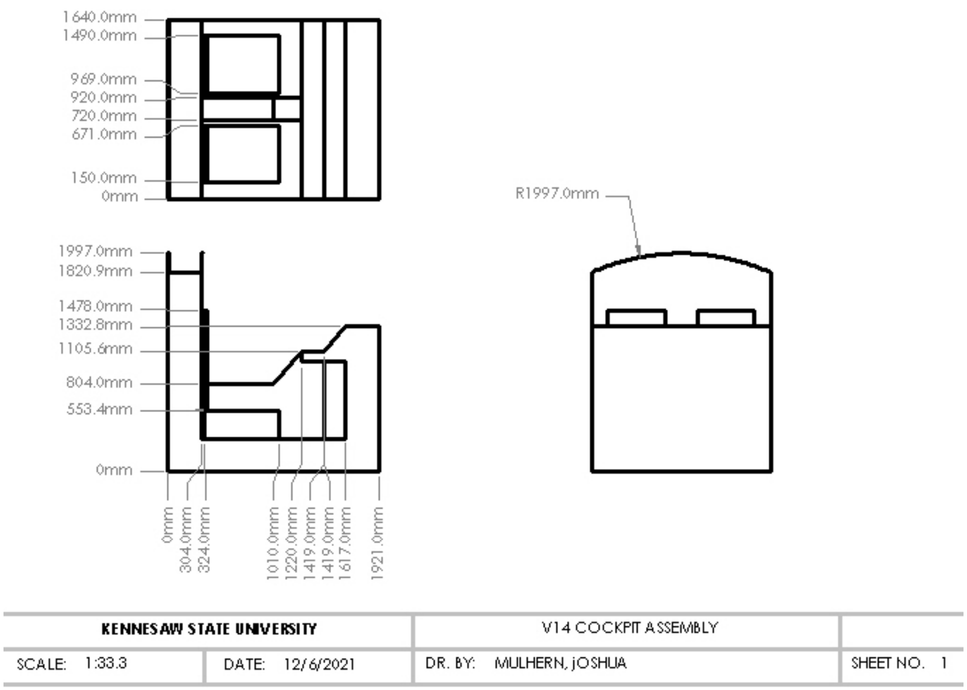


Figure 28: Cockpit assembly drawing

7.2 Internal Equipment Storage

The internal equipment storage cabinet is the location that all the internal science equipment will be store. Each case that carries equipment has its own space along with three spaces for frequently used laptops. Extra space not used by the cases and laptops can be used to store other equipment such as manual, checklists, and notebooks.

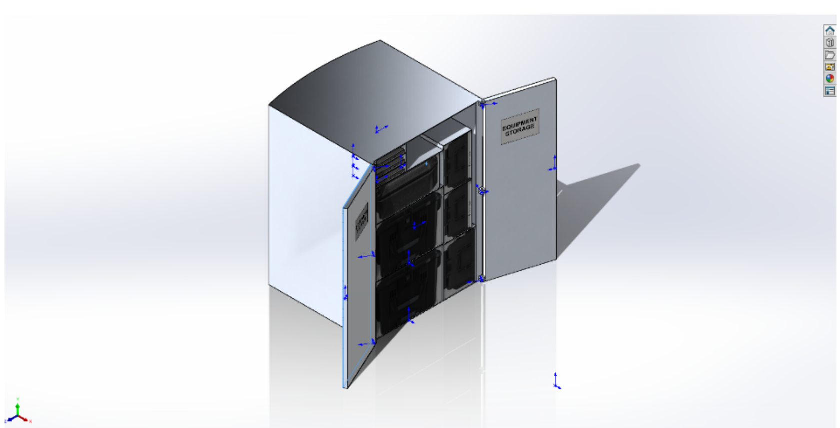


Figure 29: Equipment storage with doors open showing equipment cases neatly arranged

7.3 Food and Miscellaneous Storage

The food and miscellaneous storage cabinet is of similar size as the internal equipment storage cabinet. The food and miscellaneous storage cabinet differs by only having one shelf inside of it. There is enough space on the shelf to store every meal as mention in section 5.2.3 Under the shelf is miscellaneous storage. Anything can be stored here, from duct tape and tool to sleeping bags and replacement part.

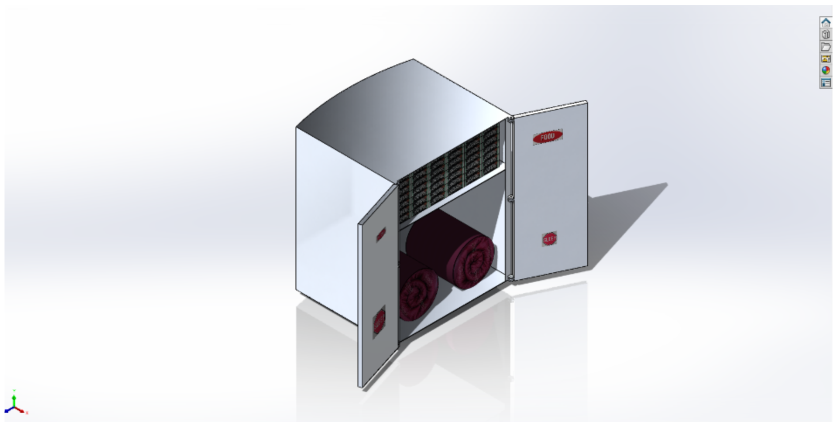


Figure 30: Food and miscellaneous storage with doors open

7.4 Bathroom

The bathroom is a stall that contains the Universal Waste Management System (UWMS). [Autrey 5] The stall itself is the same depth as the cabinets but tall enough to fit a standing person. The decision to fit the UWMS into a stall instead of in the open to save weight was due to the priority was on privacy and dignity of the astronauts instead of weight saving in this matter.

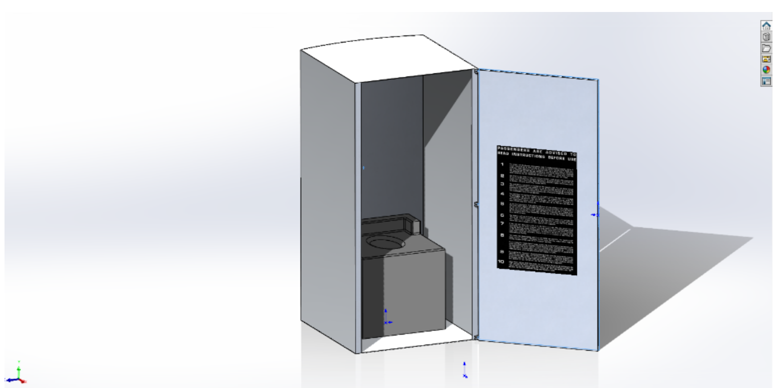


Figure 31: Bathroom assembly with door open

8 Finite Element Analysis

To see all of the pictures of the FEA's see Appendix C.

8.1.1 Version Four

Version four consisted of a flat topped 0.41mm thick pressure vessel made of aluminum 2219-T0. It had an 800mm hole for the docking port and three square windows cut out of the top face. The top and bottom frames only had two hoops and ten straight beams while the cage frame three hoops and five vertical beams.

Table 5: Version four Finite Element Analysis inputs and results

Version Four Finite Element Analysis Inputs and Results				
Force		Amount		
Acceleration		$58.86 m/s^2$		
Version Four Finite Element Analysis Results				
Analysis Name	Average stress (ksi)	Max stress (ksi)	Average Displacement (mm)	Max Displacement (mm)
Launch Forces	14.8	29.6	100.5	201

8.1.2 Version Five

Version five was the when the conical ends were added. Top and bottom frames were changed to match the new diameter. Diagonal frames were created and added. The diagonal frames at this stage consisted only two hoops and ten straight beams. The aluminum alloy was also changed to aluminum 2219-T62.

Table 6: Version five Finite Element Analysis inputs and results

Version Five Finite Element Analysis Inputs and Results		
Force	Amount	
Pressure	16.905 psi	
Version Five Finite Element Analysis Results		
Analysis Name	Average Displacement (mm)	Max Displacement (mm)
Pressure 1.15 Atm	100.5	201

8.1.3 Version Six

Changes in version six consist of a third hoop being added to the diagonal stringer one third of the way up the conical section. To reduce time between version only FEA ran was for 1.15 atmospheres of pressure

Table 7: Version six Finite Element Analysis inputs and results

Version Six Finite Element Analysis Inputs and Results				
Force		Amount		
Pressure		16.905psi		
Meshing Information				
Mesh Type		Blended Curvature-Based Mesh		
Maximum Element Size		50mm		
Minimum Element Size		10mm		
Minimum Number of Elements in a Circle		8		
Element Growth Size Ratio		3		
Version Six Finite Element Analysis Results				
Analysis Name	Average stress (ksi)	Max stress (ksi)	Average Displacement (mm)	Max Displacement (mm)
V6 Pressure 1.15 Atm	99.8	166	85.1	142

8.1.4 Version Seven

Changes in version seven consist of the diagonal frame containing four evenly spaced hoops. To reduce time between version only FEA ran was for 1.15 atmospheres of pressure

Table 8: Version seven Finite Element Analysis inputs and results

Version Seven Finite Element Analysis Inputs and Results	
Force	Amount
Pressure	16.905psi
Meshing Information	
Mesh Type	Blended Curvature-Based Mesh
Maximum Element Size	50mm
Minimum Element Size	10mm
Minimum Number of Elements in a Circle	8
Element Growth Size Ratio	3
Version Seven Finite Element Analysis Results	

Analysis Name	Average stress (ksi)	Max stress (ksi)	Average Displacement (mm)	Max Displacement (mm)
V7 Pressure 1.15 Atm	142	236	91.5	136

8.1.5 Version Eight

Changes to version eight are once again only on the diagonal frame This version see he addition of ten straight beams around the frame. To reduce the time between versions only displacement was the focus of this version's FEAs.

Table 9: Version eight Finite Element Analysis inputs and results

Version Eight Finite Element Analysis Inputs and Results		
Force	Amount	
Pressure	16.905psi	
Meshing Information		
Mesh Type	Blended Curvature-Based Mesh	
Maximum Element Size	50mm	
Minimum Element Size	10mm	
Minimum Number of Elements in a Circle	8	
Element Growth Size Ratio	3	
Version Eight Finite Element Analysis Results		
Analysis Name	Average Displacement (mm)	Max Displacement (mm)
V8 Pressure 1.15 Atm	64.3	107

8.1.6 Version Nine

Changes in this version consist of changing the pressure vessel thickness to 2mm. To reduce the time between versions only displacement was the focus of this version's FEAs.

Table 10: Version nine Finite Element Analysis inputs and results

Version Nine Finite Element Analysis Inputs and Results	
Force	Amount
Pressure	16.905psi
Meshing Information	

Mesh Type	Blended Curvature-Based Mesh	
Maximum Element Size	50mm	
Minimum Element Size	10mm	
Minimum Number of Elements in a Circle	8	
Element Growth Size Ratio	3	
Version Nine Finite Element Analysis Results		
Analysis Name	Average Displacement (mm)	Max Displacement (mm)
V9 2mm Wall Pressure 1.15 Atm	33.4	55.7

8.1.7 Version Ten

The change in this version is increasing the pressure vessel thickness to 3mm. This is also the last version that the stringer frameworks or the pressure vessel changes.

Table 11: Version ten Finite Element Analysis inputs and results

Version Ten Finite Element Analysis Inputs and Results				
Force	Amount			
Pressure	16.905psi and 29.4psi			
Acceleration	58.86m/s^2 and 88.29 m/s^2			
Meshing Information				
Mesh Type	Blended Curvature-Based Mesh			
Maximum Element Size	50mm			
Minimum Element Size	10mm			
Minimum Number of Elements in a Circle	8			
Element Growth Size Ratio	3			
Version Ten Finite Element Analysis Results				
Analysis Name	Average stress (ksi)	Max stress (ksi)	Average Displacement (mm)	Max Displacement (mm)
V10 3mm Wall Pressure 1.15 Atm	92.4	154	22.4	37.4
V10 3mm Wall Pressure 2	185	308	44.9	74.8

Atm				
V10 3mm Wall Launch Forces 6G	8.38	14	1.67	2.79
V10 3mm Wall Launch Forces 9G	12.6	21	2.51	4.18

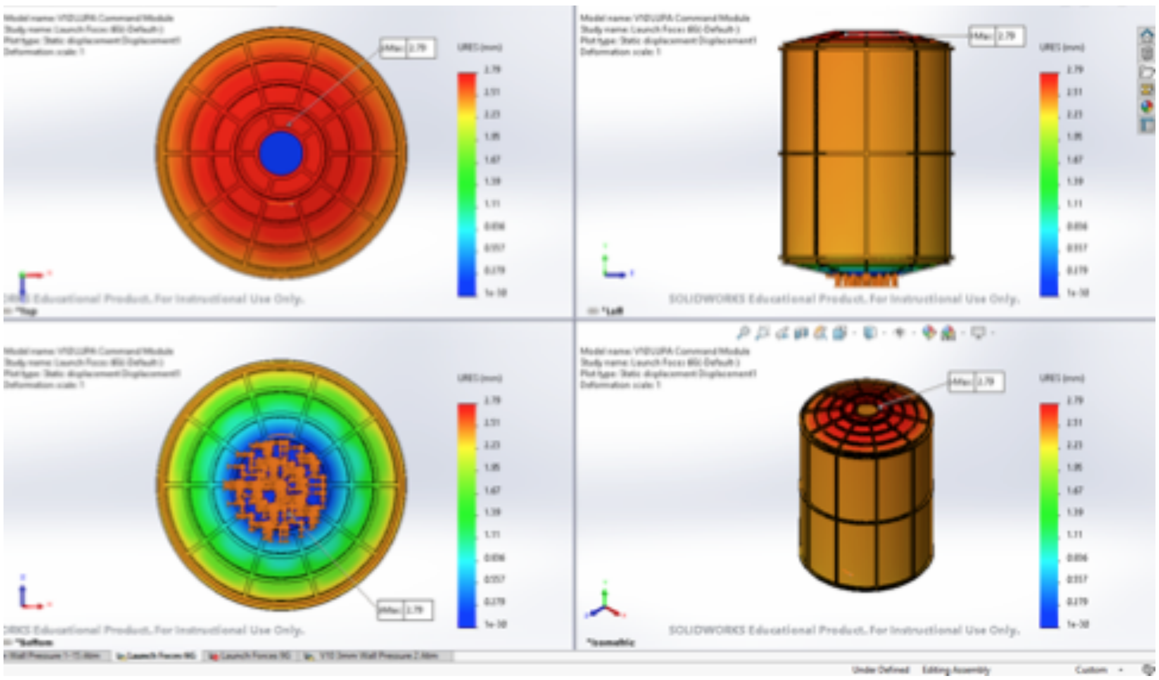


Figure 33: Version ten LUPA Command Module 6G displacement

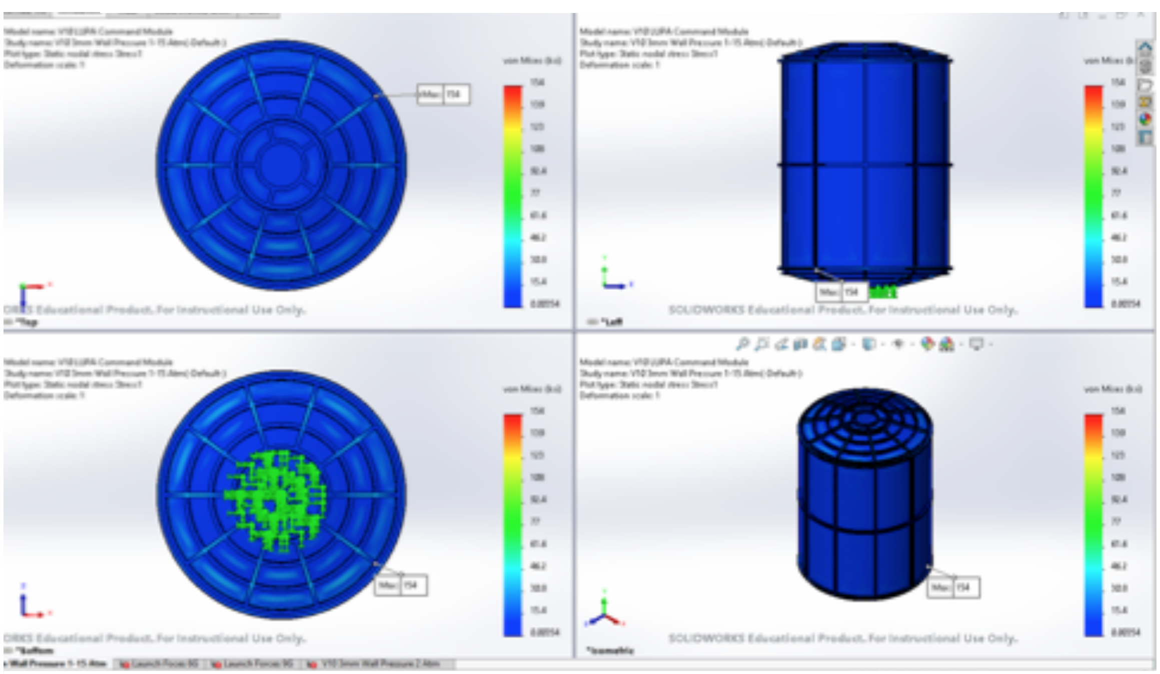


Figure 32: Version ten LUPA Command Module 1.15 Atmospheres stress FEA

8.1.8 Version 12

This version consists of the addition of the hull, the standoff frame, and the debris shield.

Table 12: Version twelve Finite Element Analysis inputs and results

Version Twelve Finite Element Analysis Inputs and Results				
Force		Amount		
Pressure				
Meshing Information				
Mesh Type		Blended Curvature-Based Mesh		
Maximum Element Size		50mm		
Minimum Element Size		10mm		
Minimum Number of Elements in a Circle		8		
Element Growth Size Ratio		3		
Version Twelve Finite Element Analysis Results				
Analysis Name	Average stress (ksi)	Max stress (ksi)	Average Displacement (mm)	Max Displacement (mm)
Launch Forces 6G	11.9	19.9	1.21	2.01
Launch Forces 9G	17.8	29.7	1.51	3.02

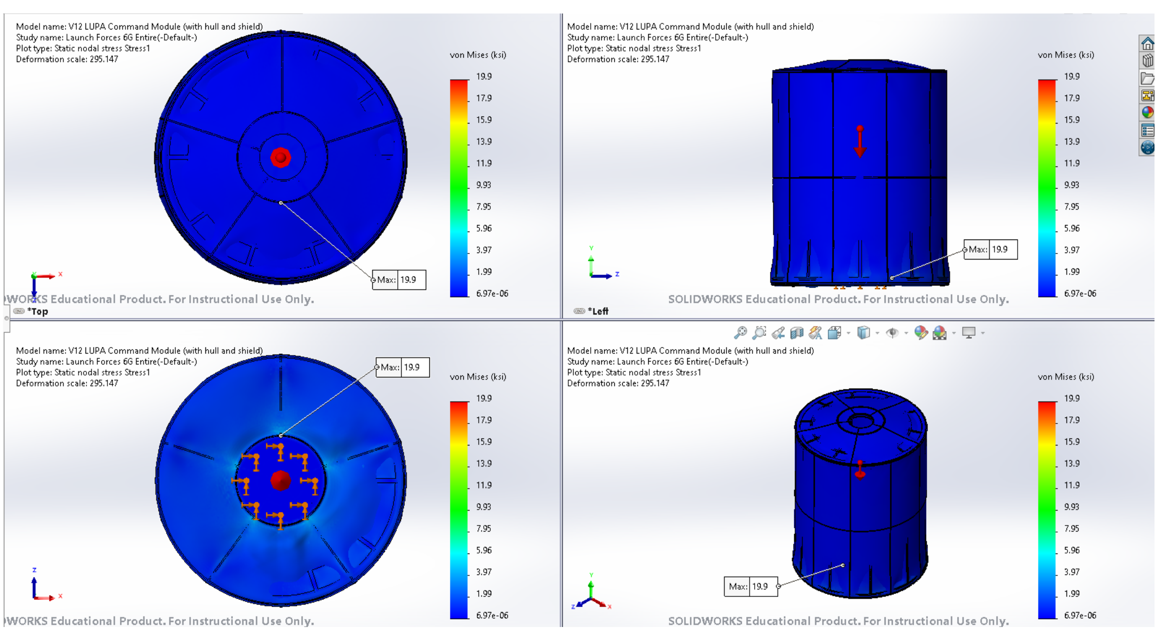


Figure 34: Version Twelve LUPA Command Module 6G Stress

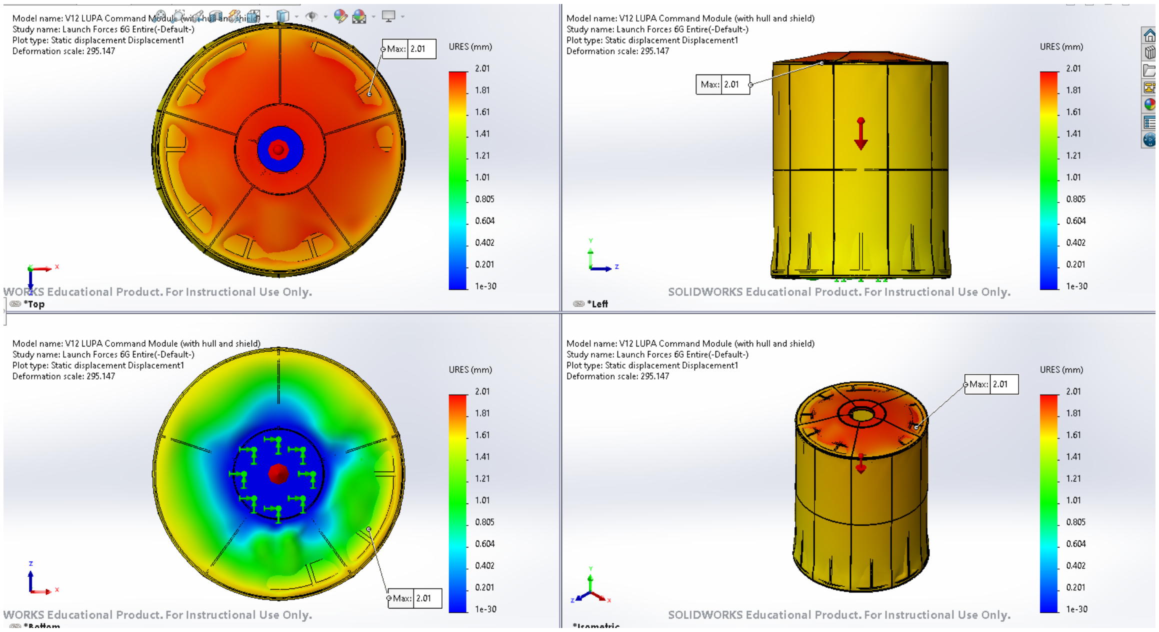


Figure 35: Version twelve LUPA Command Module 9G displacement

9 Mission Analysis

Determining the specific ΔV requirements and sequence of orbital operations were critical in ensuring that our design satisfied the core requirements of the competition. To accomplish this task, the problem was approached first through the selection of a propellant which would lead to key values which significantly impacted the mass and capabilities of the vehicle. After selecting a propellant, a corresponding engine was selected in order to ascertain parameters such as thrust and specific impulse which are necessary for the calculation of ΔV and time of flight in orbital maneuvers.

Equation 2: The ΔV Equation

$$\Delta V = \ln \frac{m_{wet}}{m_{dry}} * I_{sp} * g$$

9.1 ΔV Requirements

The first and largest major change in velocity LUPA would need to execute was during planetary capture. On the trans-Mars trajectory established by the Falcon Heavy, LUPA would be positioned to just barely enter Mars' sphere of gravitational influence. In order to enter orbit around Mars instead of flying right past it, a change in velocity dependent on the launch date needed to be calculated. This was done using "porkchop" plots of characteristic energy versus time of flight. In this plot, seen in Figure 36, a contour whose height represents orbital characteristic energy allows for the quick determination of most efficient launch dates. The most efficient launch date in 2035, seen in Figure 36 where the dotted red line representing a 200-day flight intersects the most darkly colored region of the contour, was determined to be within a three week period from the end of July to the beginning of August. As such, the formal target launch date for LUPA was set to 21 July 2035.

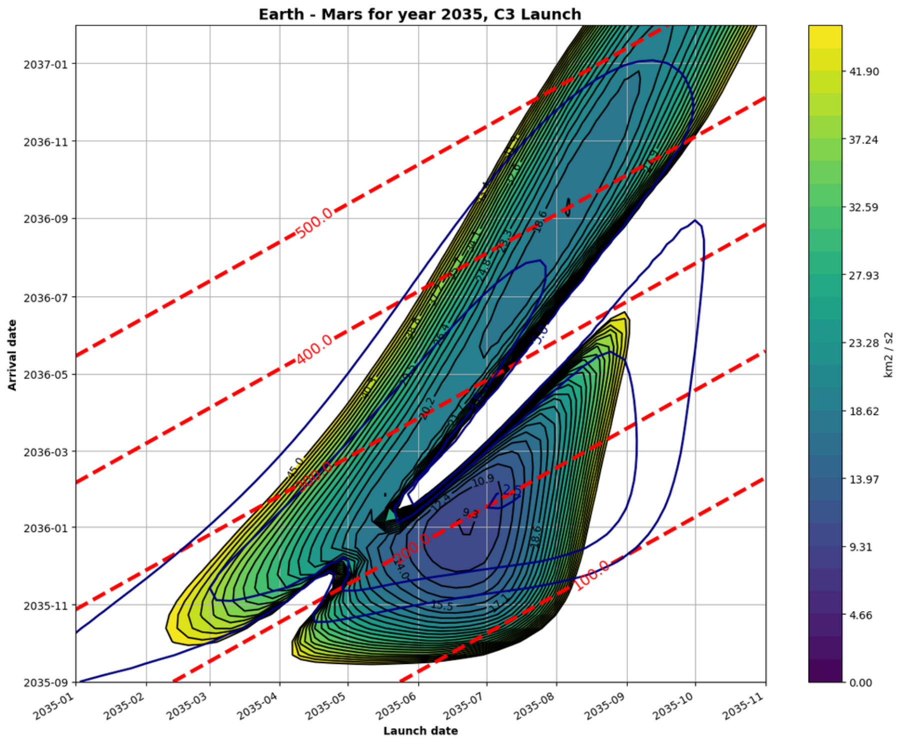


Figure 36: Porkchop plot for the 2035 Mars transfer window

By launching at this date, LUPA would only need to change its velocity by 2,600 m/s to enter Mars orbit. After being successfully captured by Mars' gravity, a secondary maneuver to place LUPA in the 5-sol parking orbit expected by the eventually arriving crew aboard the DST will cost an additional 560 m/s. Since the 5-sol orbit is most likely highly elliptical, based on NASA documentation such as that illustrated in Figure 37 below, the transfer is calculated by using the vis-viva equation. This is the same equation underpinning the calculations used in circular Hohmann transfers, however circular transfers are much simpler.

Equation 3: The Vis-Viva Equation

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

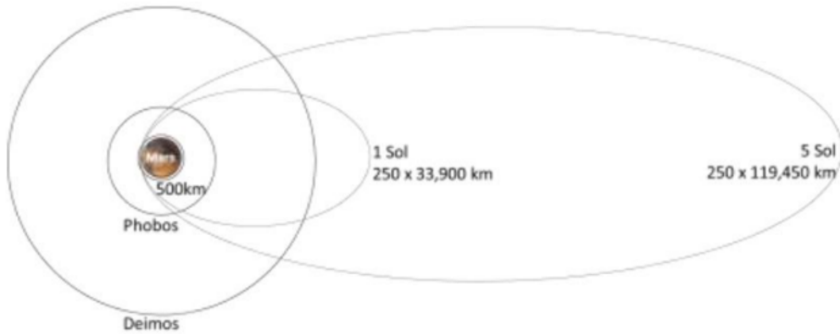


Figure 37: Diagram showing relative orbital altitudes

To calculate the ΔV required to carry out the various rendezvous operations performed by LUPA, each segment of the sortie was considered to be a standard, circular Hohmann transfer. Since the orbits of both Phobos and Deimos are only a few hours, the time necessary to wait for ideal alignment was considered negligible.

Table 13: Orbital maneuver ΔV and time budget

	ΔV (m/s)	TOF (Days)
Mars capture	2600	
Mars capture to 5-sol	559.9	
5-sol to Phobos		2.93
Raise Periapsis	242.8	
Circularize at Phobos	709.5	
Phobos to Deimos		0.37
Raise Apoapsis	409.1	
Circularize at Deimos	324.6	
Deimos to 5-sol		3.55
Lower Periapsis	115.3	
Raise Periapsis to 5-sol	703.1	
Total	4989.1	6.25

From the calculated values tabulated above, it can be seen that LUPA's total ΔV was just under 5,000 m/s which was sized to 5,500 m/s in subsequent sizing calculations for redundancy. This requirement was used in accordance with the properties of the hydrazine/nitrogen tetroxide propulsion system selected in section 5.1, Propulsion, to calculate the propulsion mass values and subsequent remaining "dry" mass budget discussed further in the following section.

10 Cost Analysis

10.1 Bill of Materials

1. Elements of Life-Cycle Cost

The element of life-cycle cost is the approach that assess the total cost of the spacecraft over its life cycle. It consists of the initial Research, development, test, and evaluation, Production construction cost, ground support equipment and initial spares, special construction, operation costs, and maintenance cost. Research, development, test, and evaluation (RDT&E) consist of basic research, applied research, advanced technology development, advanced component development and prototypes, system development and demonstration, RDT&E management support and operational system development. The production stage consists of the space production, engine production, and avionics production. While the ground support equipment and initial spares consists of facilities, equipment, software, logistics, system engineering, product assurance, management, communication, flight and ground software, integration, and test. Operations and maintenance consist of crew personnel, maintenance, recurring, depot, insurance, indirect costs, and depreciation.

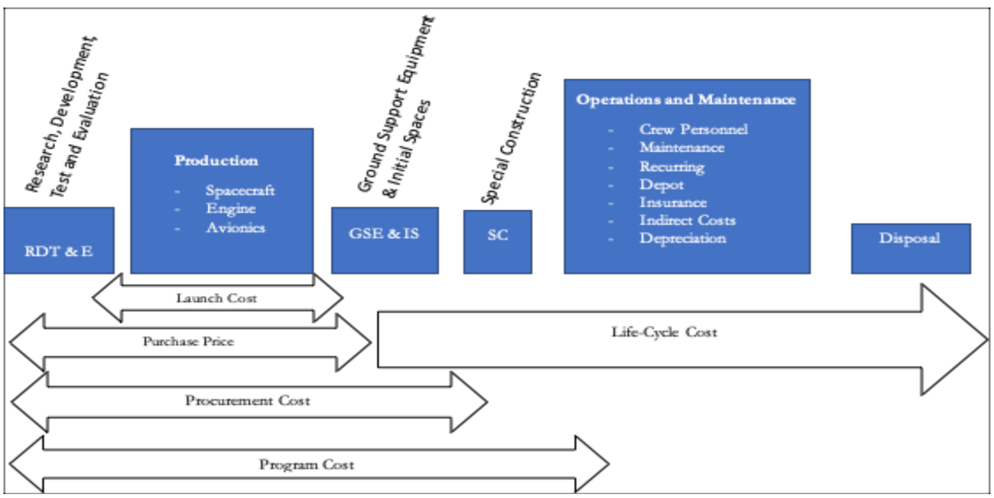


Figure 38 Element of Life-Cycle Cost

2. THE COST ANALYSIS FOR THE MISSION

NASA allotted a budget of not more than $1 billon for the project. The most expensive segment is the ground segment which cost about $528,415,898.58. Each element has a considerable contribution as shown on the chart below. Space system has the lowest contribution. Cost estimation process was done using parametric estimation from Aircraft Design: A Conceptual Approach textbook. Operation and maintenance costs were determined from assumptions as to how the spacecraft will be operated. Launch vehicle cost about $136,352,007.74, ground segment cost about $528,415,898.58, space system cost about $24,718,976.60, operations and maintenance cost about $136,352,660.74, and advanced technologies cost about $84,104,976.66.

Table 14: Cost Analysis

System Segment	Subsystem	Cost (Millions)	
Launch Vehicle		$	138,576,007.17
	Falcon Heavy	$	112,992,744.31
	Launch Operations	$	25,583,262.86
Ground Segment		$	528,415,898.58
	Facilities	$	25,135,555.76
	Equipment	$	113,110,000.92
	Software	$	139,641,976.45
	Logistics	$	20,946,296.47
	System Engineering	$	41,892,592.93
	Product Assurance	$	20,946,296.47
	Management	$	25,135,555.76
	Communication	$	1,965,647.36
	Flight and Ground Software	$	139,641,976.45
	Intergration and Test	$	33,514,074.35
Space Systems		$	24,718,976.60
	Proplusion	$	1,553,192.03
	Structure	$	1,124,945.67
	Payload	$	4,261,157.83
	Ground Support	$	703,091.04
	Thermal	$	245,016.58
	Integraion, Assembly, and test	$	1,480,752.35
	EPS	$	21,305.79
	Program Level	$	2,439,512.86
	Command and Data Handling	$	1,691,679.66
	Attitude Determination	$	2,053,878.08
	Spacecraft	$	9,144,444.71
	Electrical Power System	$	6,658,059.12
	Telemetry	$	1,380,615.14
	Lauch and Operation	$	649,826.57
Operations and Maintenance		$	136,352,660.74
	Maintenance	$	55,577,506.63
	Government Labor	$	117,256.62
	Contract Labor	$	80,657,897.49
Advanced Technologies		$	84,104,976.66
	Solar Panels	$	27,022,321.40
	Capusle	$	19,667,133.32
	Torpor	$	37,415,521.93
TOTAL		$	912,168,519.73

Ground Segment budgets were allotted 4% for facilities, 20% for equipment, 25% for software, 20% of the budget for logistics, 4% for system engineering, 4% for product assurance, 4% management, 1% for communication, 25% for flight and ground software, 6% for integration, and test

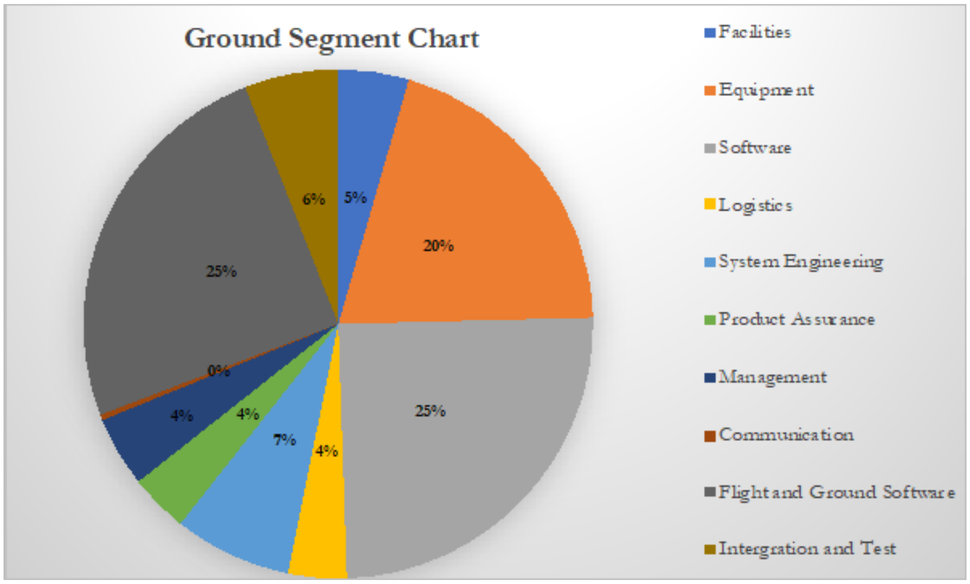


Figure 39 Ground Segment

The space system consists of payload, spacecraft, structure, thermal, electrical power system, propulsion, program level, EPS, launch and operation, ground support, telemetry, electric power system, command and data handling, attitude, and determination. the highest budget for space system segment will be spent on structure because is 27%, 2% ground support, 2% launch and operation, 20% electric power system, 4% telemetry, 1 and 1% of the budget.

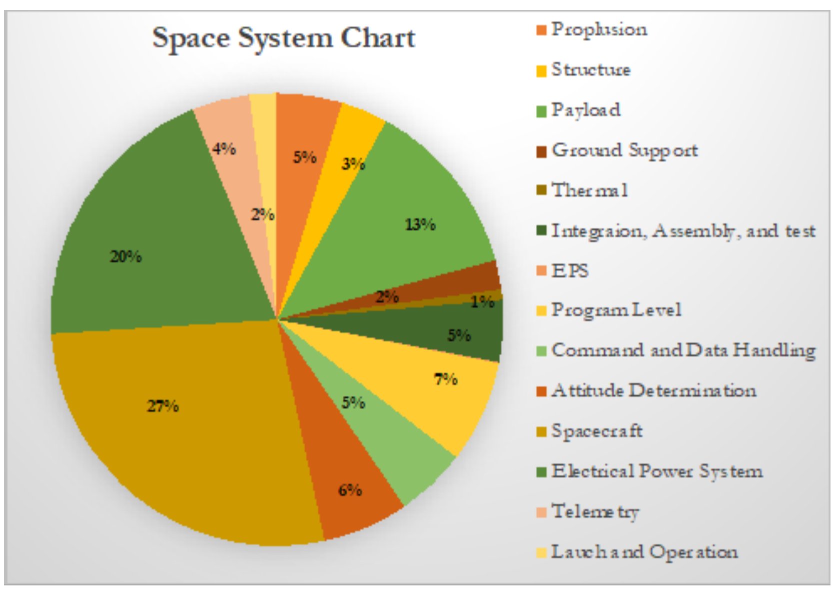


Figure 40 Space System Chart

Advanced technologies segment budget 35% of the budget is allotted to torpor, 22% for space station Bigelow module, 25% for solar panels, and capsule 18%.

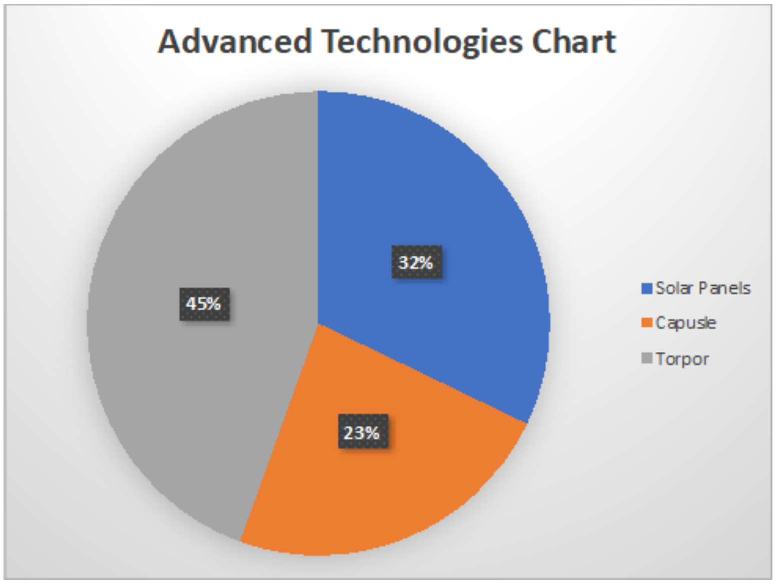


Figure 41 Advanced Technologies Chart

10.2 Ground Operations

Ground operations are operation that will successfully complete on earth before the launching stage to ensure that the vehicle is completely built to withstand all strain and stress. Also ensure that the crew member successful accomplish their mission with less health hazard. They are various stages involved in ground operations. The timeline for the entire ground duration of the mission can be seen in figure 8.3 below

- RDTE will take 4 years (January 2022 – January 2026) to complete research, development, test, and evaluation
- Production involves 2 stages which are vehicle production launch and vehicle production assembly. Vehicle production launch will take two and half years (January 2026 – January 2031) and vehicle production assembly will take two and half years (January 2022 – January 2026) to complete.
- Operation review will take one and a half years (January 2031 – January 2032) to complete
- Vehicle readiness review will take a year and half (January 2032 – January 2033) to complete the evaluation to ensure that all components of the vehicle and vehicle are functioning well.
- Launch is the last stage on earth which is schedule to take place between 21st June – 5th July 2035.

There is 2 years interval between the vehicle readiness review and launch date between in case we need to reschedule, delay in supply or errors we will have time to fix and still stay on schedule.

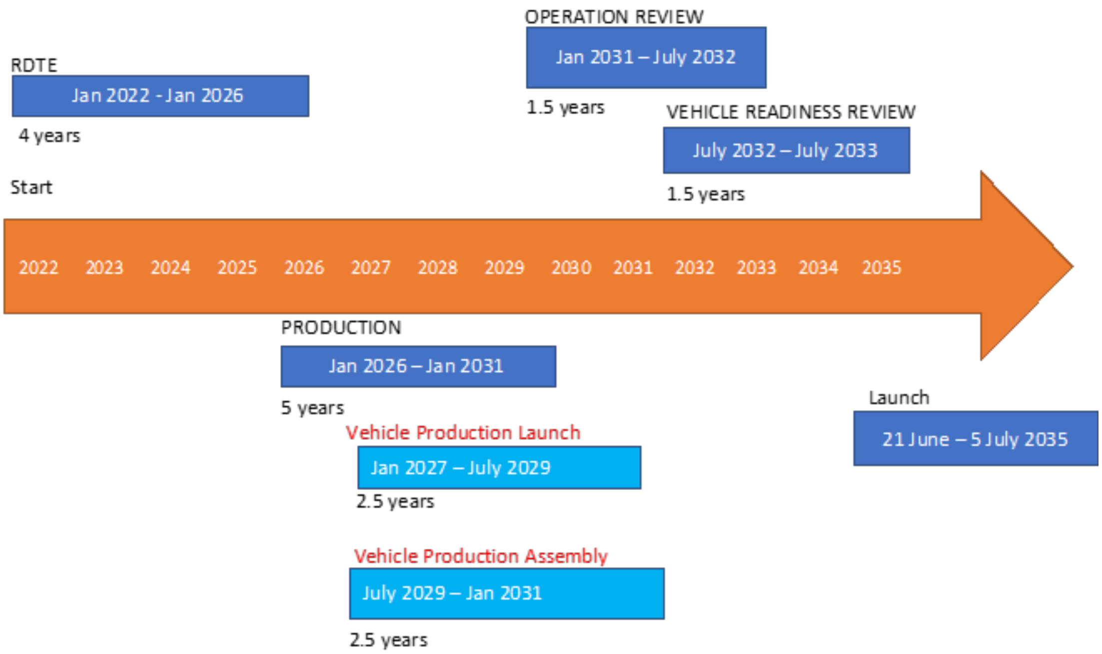


Figure 42 Ground Operation Timeline

10.3 Total Budget

Cost estimation was based upon detailed assessment of the actual design, test, and production of the spacecraft. Cost estimation was mostly statistical during conceptual design. The total estimated cost for the project will cost $912,168,519.73. Advanced technologies cost about $84,104,976.66, space systems cost about $24,718,976.60, launch vehicle cost about $138,576,007.17, ground segment cost about $528,415,898.58, and operations and maintenance cost about $136,352,660.74.

Table 15: Cost Estimation

COST ESTIMATION		
Segements		
Advanced Technologies	$	84,104,976.66
Space Systems	$	24,718,976.60
Launch Vehicle	$	138,576,007.17
Ground Segment	$	528,415,898.58
Operations and Maintenance	$	136,352,660.74
Total	$	912,168,519.73

11 Risk Management

Risk management is used to identify and understand the possible risks impacts on operational processes and make decisions on operational processes and act to manage potential undesired effects. Risk management helps to avoid loss of vehicle, loss of crew member, incomplete mission, and loss of sample. Lunar and mars mission required critical planning and consideration to ensure that the mission is completed. The main purpose of risk management is to ensure that all crew members are safe and healthy. Table 9.1 lists the risk levels and likelihood levels and table 9.2 list the possible risk for the mission with possible solutions.

Table 16: Risk Management Key for Table 7 Below

LIKELIHOOD	
Type	Level
Less	1
Medium	2
High	3
RISK	
Type	Level
Minor	1
Medium	2
Moderate	3
Critical	4
Catastrophic	5

Table 17: Risk Management and Possible Solution Table

	Description	Likelihood	Risk	Solution
1	Engine Failure	2	1	Use redundant engine
2	Propulsion Malfuction	2	1	Test the tank and fitting systems on ground completely and ensure that the engine have over 99% reliability
3	Products Delivery Delays	2	3	Reschedule delivery date
4	Antenna Failure	1	2	Use redundant antenna and install new path to transmit signals to Deep Space Gateway
5	Power Failure	3	1	Use redundant thermal subsystem
6	Landing Gear Failure	2	2	Use extractor and extract samples
7	Missing Planned Lauched Date	3	4	Reschedule launch
8	Battery Shortage	1	3	Use battery size that will substain sufficient power
9	Corrosion	3	4	Use shield layer over the structure
10	Cost Overrun	1	4	Use money from the 10% reserve from the budget
11	Schedule Overrun	1	4	Reschedule launch
12	Crew Health Care	2	4	Radiation shielding, santization
13	Loss of Attitude	1	5	Use redundant thurster
14	Solar Panel Failure	2	3	Use high margin EPS sizing
15	Processor Unit Failure	1	4	Use alternative or redundant instruments
16	Environment Impact	2	2	Use alternative design for worst case
17	Landing leg fails to Separate	2	2	Test in NASA Deep Space Gateway
18	Structure Fatigue	2	5	Delay lanuch, rebuild the structure
19	High Concentraion of Oxygen	2	1	Use pressure senor
20	Electronic Instrument Error	1	3	Use surplus wires, switches and components
21	Loss of communication	1	2	Use high margin data transmission
22	Mass overrun	2	5	Set mass margin
23	Damage encured during unloading operation	2	2	Use redundant and alternative methods
24	Equipment ignition	2	4	Fire detection, vaccum, EVA suits
25	Crack formation	1	4	Mantienance and monitoring
26	Vechile Component and Subsystem overheating	2	2	Use multi layer insulation
27	Soft Landing Failure	2	3	Repair the vechicle system

The risk management chart helps to calculate the possibility of each risk by multiplying the likelihood and impact of each risk element for easier examination to know on what element to focus the most. The elements on the red zone on chart 9.1 below need to be monitored, redesign or inspect often to avoid any catastrophic before, during or after the mission. To ensure all crew members and the mission arrive back to earth safety. The green zone has less likelihood of occurring while the yellow zone has medium chance of occurring and must be avoided.

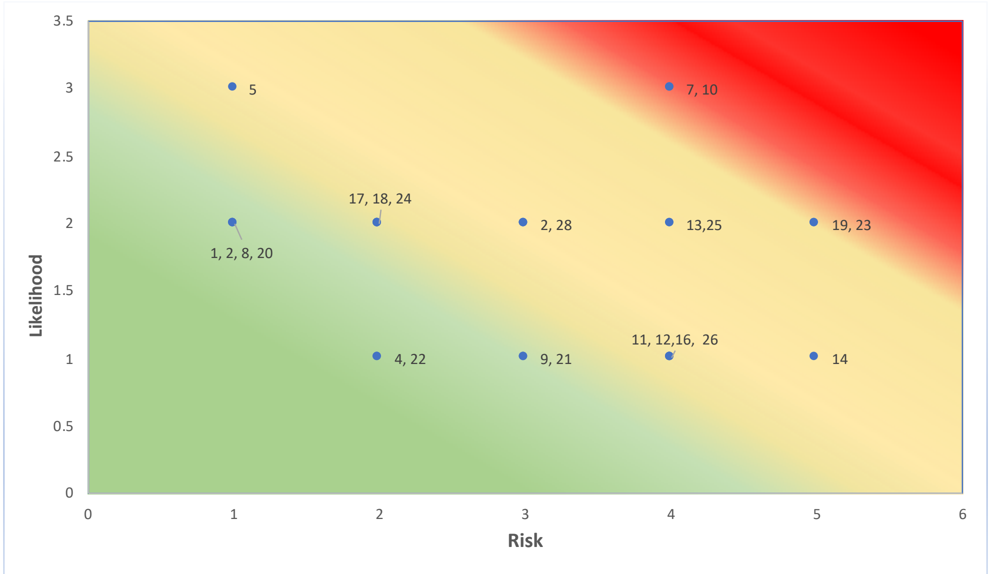


Figure 43 Risk Management Chart

12 Acknowledgements

We would like to acknowledge several people who have indirectly contributed to the progress outlined in this report by means of sharing advice and expertise. In no particular order, we thank:

- Benjamin Cardenas, Geology Professor at Pennsylvania State University
  - For advice concerning the selection of scientific equipment
- James Kasting, Geology Professor at Pennsylvania State University
  - For possible scientific objective concerning the Martian moons Phobos and Deimos
- Matthew Bartow, Payload Operations Controller at ISS Mission Control
  - For points of advice concerning the selection of scientific equipment
- Dr. Shawn Cruzen and the rest of the staff at Columbus State University's Coca-Cola Space Science Center
  - For their advice and thoughts on our mission from the perspective of astrophysicists, enthusiasts, and educators.
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- Eric Zito for his help with determining proper load and limits for the structure of the spacecraft.

12.1 Individual Contributions

12.1.1 Bo Lewis

Building off research and preliminary calculations performed by Alex, Bo performed propellant trade studies and selected a hydrazine propulsion system and a corresponding engine. Bo was responsible for writing the Propulsion System, Orbital Mechanics, and Sizing Calculations sections and created all 3D renders of the spacecraft contained within the report.

12.1.2 Shannon Kavanagh

Shannon is responsible for the researching of external scientific equipment including the sample collection mechanism, detailed in the Scientific Equipment and Sample Collection Methods section.

Research and examination of past present and future lunar, Martian, and satellite vehicles.

Mechanisms of sample manipulation and retrieval, arm and instrument research.

Scientific suite and navigational aids, data collection and testing, external laboratory equipment.

12.1.3 Alex Odinamba

In addition to preliminary research towards propellants, Alex Odinamba is responsible for the Mission Logistics and Preliminary Budget Estimation section as well as its respective subsections and comprising figures. Determined possible risk factors with solutions.

12.1.4 Joshua Mulhern

Wrote the foodstuff, Structure (except for Landing Legs), Internal Layout, Finite Element Analysis sections. Generated and refined SolidWorks models and assemblies for the structural support of the spacecraft. Determined the internal layout and created and refined SolidWorks models and assemblies for it. Created, ran, and analyzed forty Finite Element Analysis focus on the structural stability of the spacecraft. Researched and examined possibilities to accomplish the above. Fixed formatting errors caused by original submission.

12.1.5 Additional Contributions

We would like to acknowledge several people who have indirectly contributed to the progress outlined in this report by means of sharing advice and expertise. In no particular order, we thank:

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- Eric Zito, Aeronautical Engineer, senior structural analyst at Lockheed Martin Marietta for his help with determining proper load and limits for the structure of the spacecraft.

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14.1.1 Bo Lewis

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15 Appendix

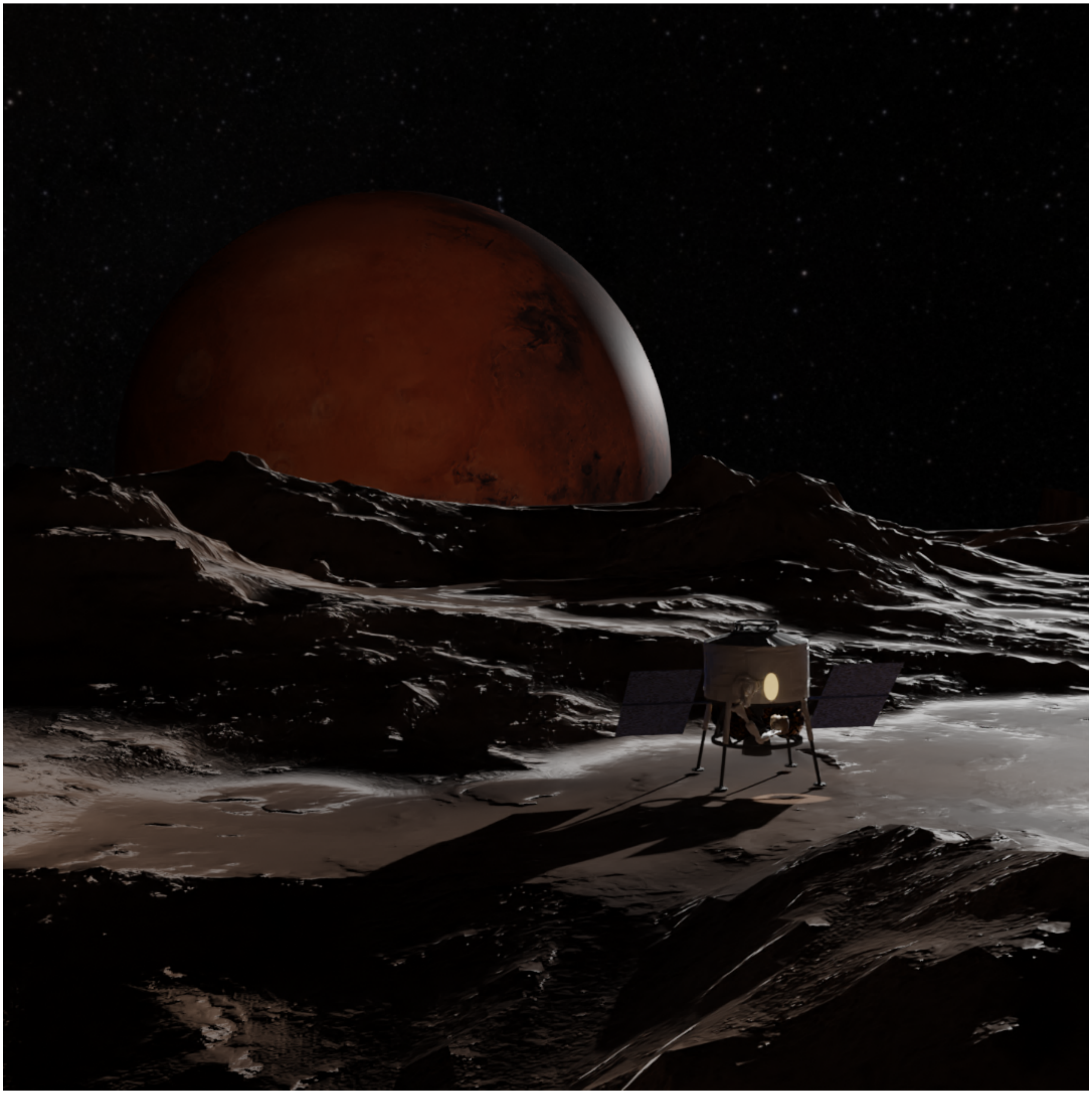


Figure 44: Render of LUPA on the surface of Phobos

Table 18: AIAA Competition Design Requirements and Constraints

- Design an Exploration Excursion Vehicle (EEV) for the Martian Moons: Phobos and Deimos
    - The EEV should have the capability to support 2 crew members to visit both Martian moons
    - The total mission shall not exceed 30 days, including transit time from the Deep Space Transport (DST) vehicle to the destination and back.
    - The EEV should be able to support sample retrieval from each destination, with a minimum sample retrieval mass of 50 kg from each moon.
    - The 2 crew member will remain inside the EEV during the mission, with no planned EVA capability
    - The team can decide to plan for a single sortie to visit both moons, or multiple sorties from the DST, as long as the total duration not exceed 30 days
- Research and define appropriate scientific objectives for the crew to during the mission sortie, to include the sample retrieval at the destination
    - The EEV should have the ability for the 2 crew members to conduct exploration of the moons to produce significant scientific understanding of the moons.
    - These scientific objectives should advance our knowledge of both moons and improve our capability to explore future destinations across the solar system.
    - Describe scientific experiment equipment that are necessary to achieve these scientific goals
    - Up to 200kg of science equipment can be delivered to the EEV with the crew on the DST, but they are limited to what the crew can carry into the EEV through the pressurized tunnel
    - Describe the sample retrieval mechanism and how the samples will be stored during the sortie and how the sample will be transferred to the DST for the return trip to Earth. The sample must be quarantined from the crew until Earth arrival for scientific study
- Design and define the mission operations, including orbit transfer, station keeping, and other maneuvers necessary for mission sortie
    - The EEV shall autonomously dock with the DST, and 2 crew will transfer into the EEV to begin the mission sortie
    - Discuss the mission modes and maneuvers required to complete the roundtrip missions to visit both Martian moons
    - Discuss the time and operation required at each destination to support the science objective as defined by the team
- Describe in detail how the vehicle will be deployed to Mars in preparation for the crew arrival.
    - Assume the Crew arrives in a DST vehicle in a Mars 5-sol parking orbit on January 1, 2040, the EEV must already be in 5-sol orbit awaiting for Crew arrival before this date
    - Discuss the launch opportunity for the EEV and the propulsion system required to deliver the EEV to Mars and the interplanetary trajectory for the EEV
    - Describe the selection of launch vehicle and the selection process that led the team to the decision
- Perform trade studies on vehicle system options at the system and subsystem level to demonstrate the fitness of the chosen vehicle design. It is highly desirable to use technologies that are already demonstrated on previous programs or currently in the NASA technology development portfolio

- Discuss selection of subsystem components and the values of each of the selection and how the design requirements drove the selection of the subsystem
- The cost for the vehicle shall not exceed $1 Billion US Dollar (in FY21), including the launch cost.

16 Appendix B Email from Eric Zito

From: "Zito, Erik"
Subject: RE: Quote for Project
Date: December 2, 2021 at 9:05:41 AM EST
To:

Joshua,

When performing a structures analysis we apply factors for various situations. Many times these factors are Program specific, and are agreed upon between the customer and supplier prior to contract award. Some of these factors may include:

Design Limit Load Factor
Ultimate Factor
Pressure Factor
Proof Pressure Factor
Fitting Factor

Design Limit Factor - This is a factor applied to limit load (loading the structure would be expected to incur during normal operation). Across the industry this number is generally on the order of 15% higher than limit (1.15). The design limit stresses of the part are then compared against the yield properties of the material (Fty, Fcy, etc.). What this means is, we would not expect to see detrimental deformation or plastic yielding of the part up to 15% greater than the structures normal operational loading.

Ultimate Factor - This is also a factor applied to limit load. Across the industry this number is generally on the order of 50% (1.5) for manned vehicles, or 20% (1.2) for unmanned vehicles. After applying the ultimate factor, we compare the ultimate internal stresses of the part to the ultimate material allowables (Ftu, Fcu, etc.). This is an indication that catastrophic failure of the part does not occur up 150% of the normal operational load. So while the structure will likely incur permanent set and excessive deformation up to this point, the part should continue to hold together in a linear analysis scenario.

Pressure Factor and Proof Factor - These factors vary widely from Program to Program and generally based upon the operational environment of the structure and what failure of the structure would mean for the mission (1.2 if low criticality - 2.0 if high criticality).

Fitting Factor - These factors are applied because fittings are generally used along single point of failure load paths and are generally essential for critical function of the structure. In my experience this factor is on the order of 15% and is used in conjunction with your Program specific ultimate or limit loading factors.

My recommendation was, if there is no requirement to have sharp corners or flat sidewalls, that the pressure vessel be a cylinder with domed ends. The benefit is you significantly reduce points of stress concentration at the welds, which are already reduced strength due to the heat effected zone, and domed/cylindrical structure is more efficient in carrying pressure. If a vessel with straight walls is pressurized, the straight wall must then carry the vessel pressure in out-of-plane bending to the corners or joints. This applies end moments at the corners or joints that are not easily reacted without additional structure (gussets/stiffeners). Conversely a cylindrical and domed pressure vessel carries the pressure in circumferential stress and longitudinal stress. This allows for end moments to not pile up at the corners and a more uniformly loaded structure.

I hope this helps. Please let me know if you have any additional questions.

Thank you,

Erik Zito

Aeronautical Engineer, Stf.
Lockheed Martin

17 Appendix C Finite Element Analysis Pictures

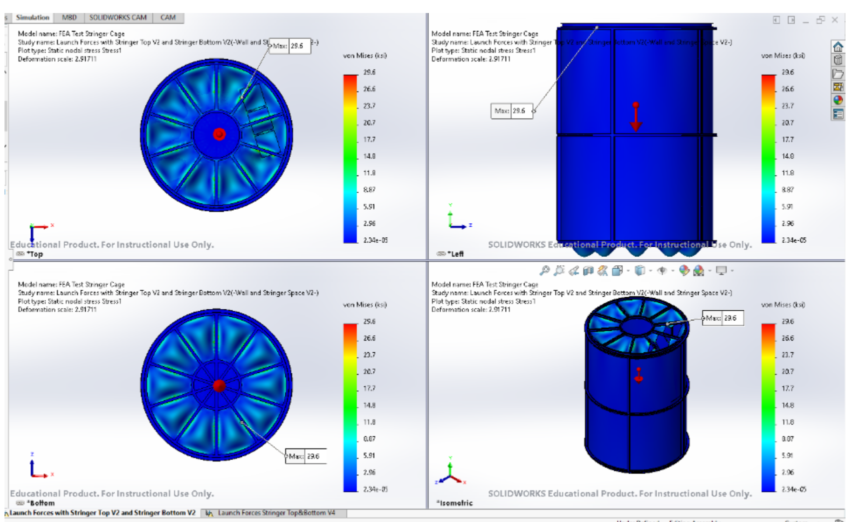


Figure 45: V4 LUPA Command Module Launch Forces 6G Stress FEA

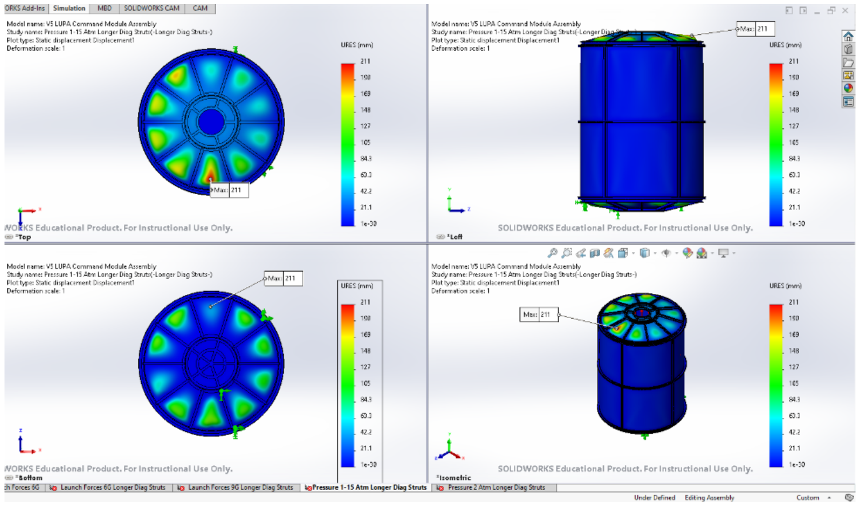


Figure 46: V5 LUPA Command Module Pressure 1.15 Atm Displacement FEA

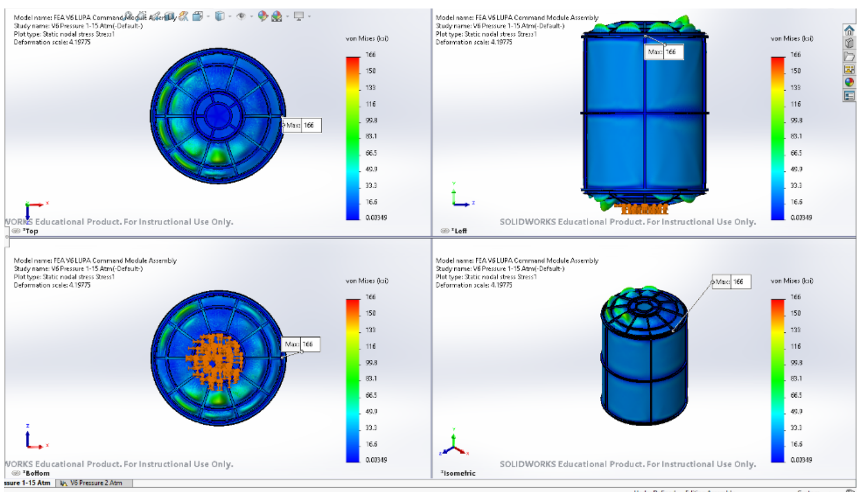


Figure 47: V6 Command Module 0.41mm Wall Pressure 1.15 Atm Stress FEA

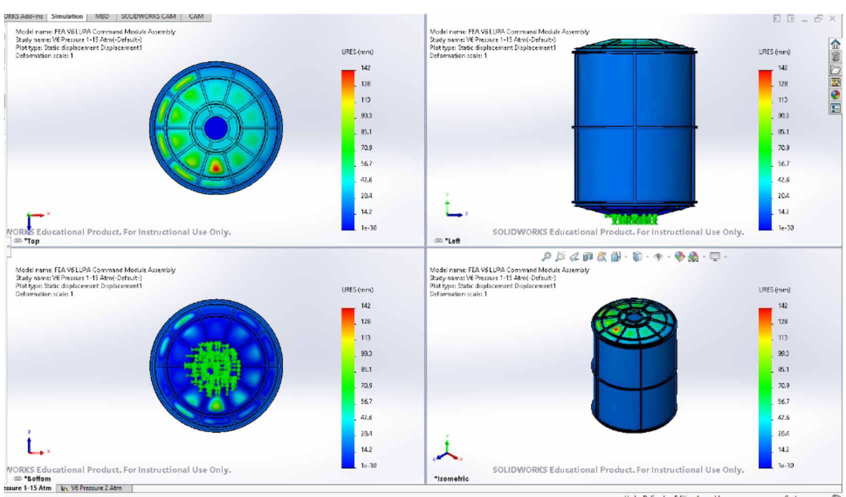


Figure 48: V6 LUPA Command Module 0.41mm Wall Pressure 1.15 Atm Displacement FEA

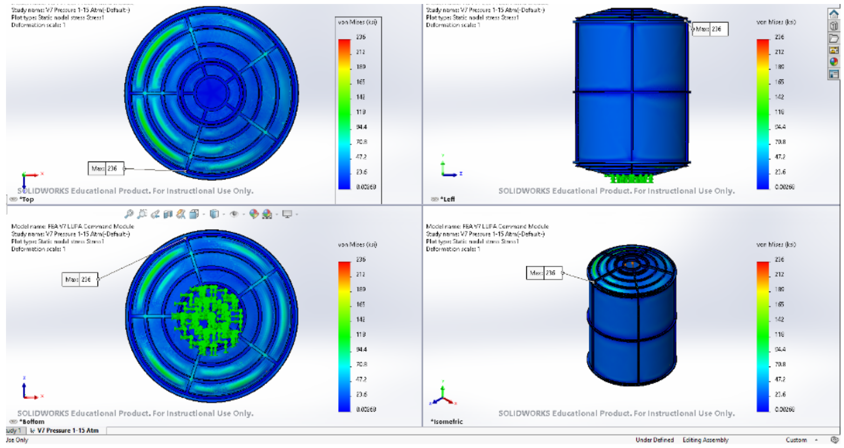


Figure 49: V7 LUPA Comand Module Pressure 1.15 Atm Stress FEA

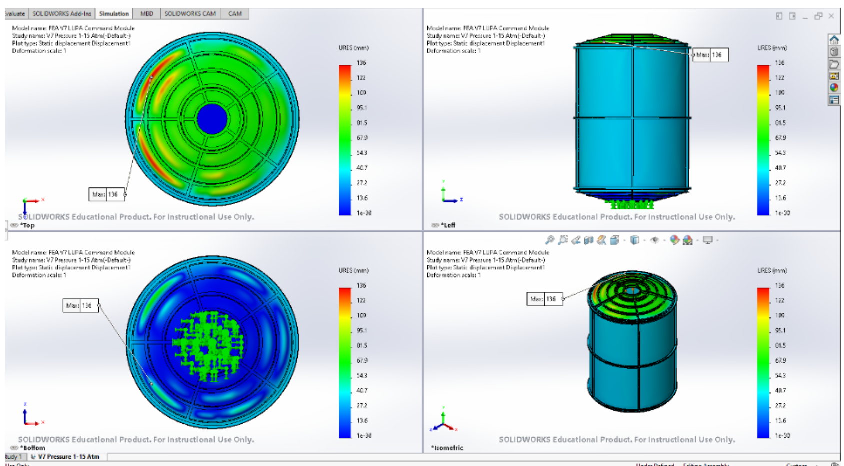


Figure 50: V7 LUPA Command Module Pressure 1.15 Atm Displacement FEA

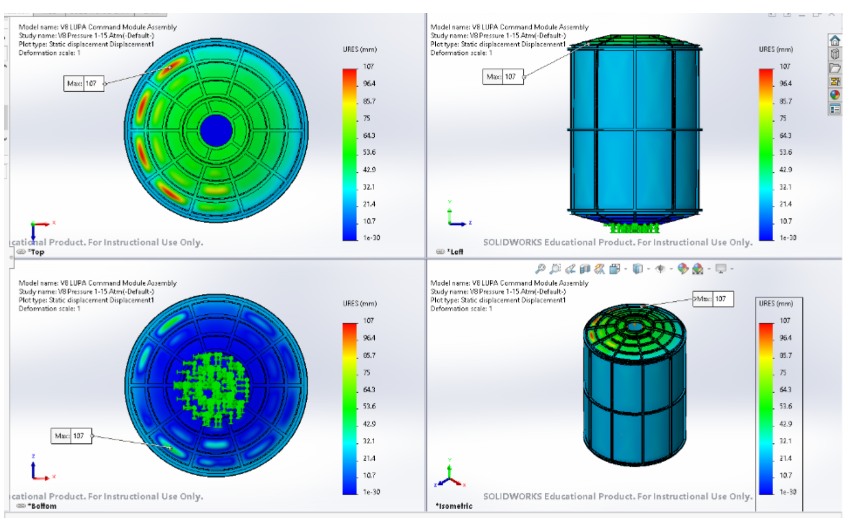


Figure 51: V8 Lupa Command Module Pressure 1.15 Atm Displacement FEA

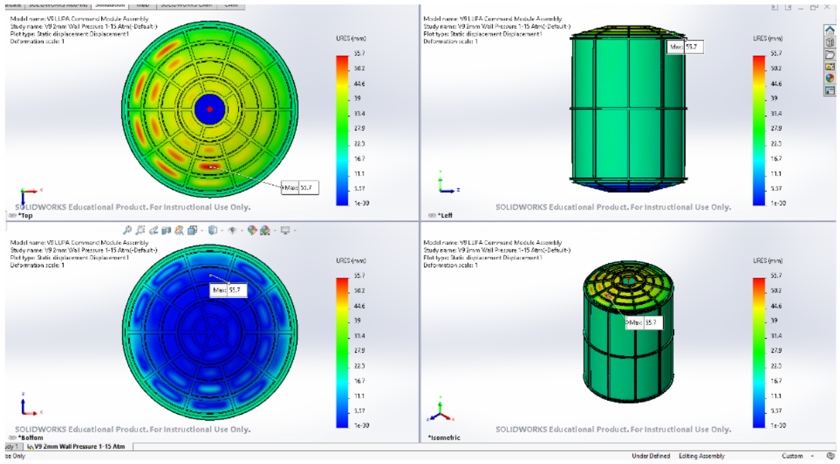


Figure 52: V9 LUPA Command Module Pressure 1.15 Atm Displacement FEA

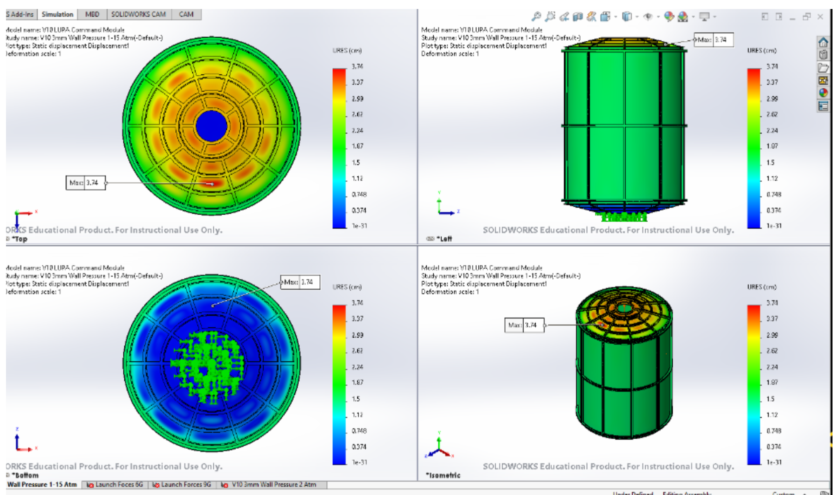


Figure 53: V10 LEPA Command Module 3mm Wall Pressure 1.15 Atm Displacement FEA

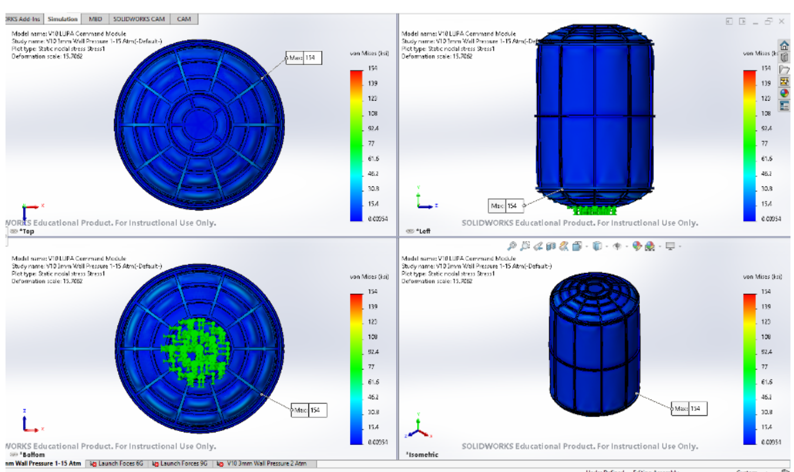


Figure 54: V10 LUPA Command Module 3mm Wall Pressure 1.15 Atm Stress FEA

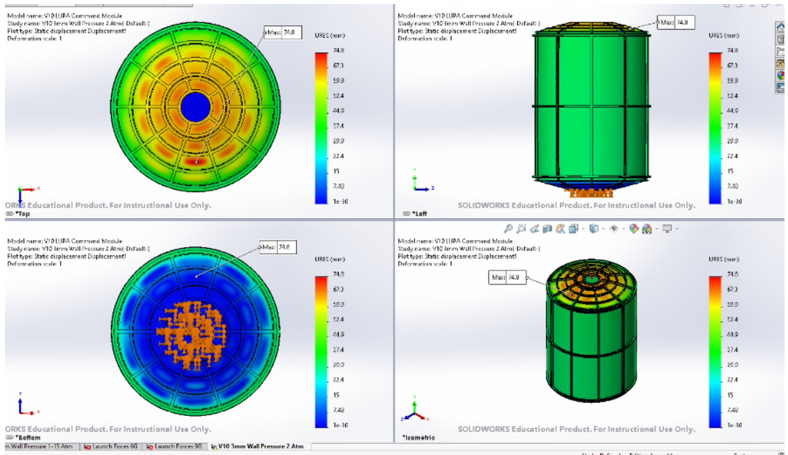


Figure 55: V10 LUPA Command Module 3mm Wall Pressure 2 Atm Displacement FEA

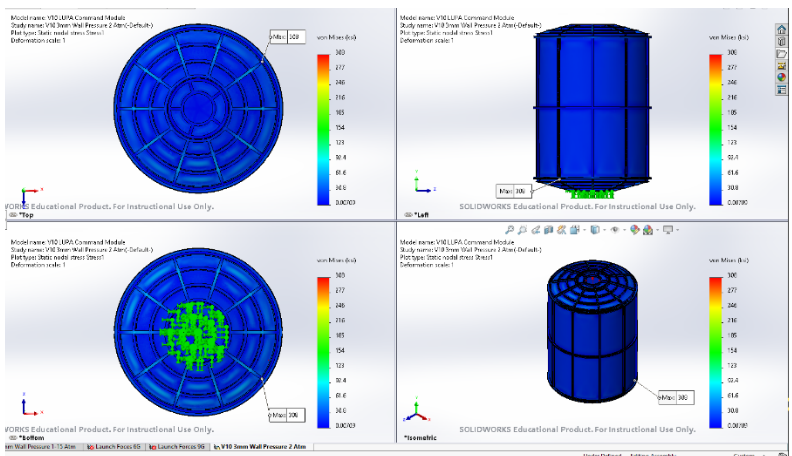


Figure 56: V10 LUPA Command Module 3mm Wall Pressure 2 Atm Stress FEA

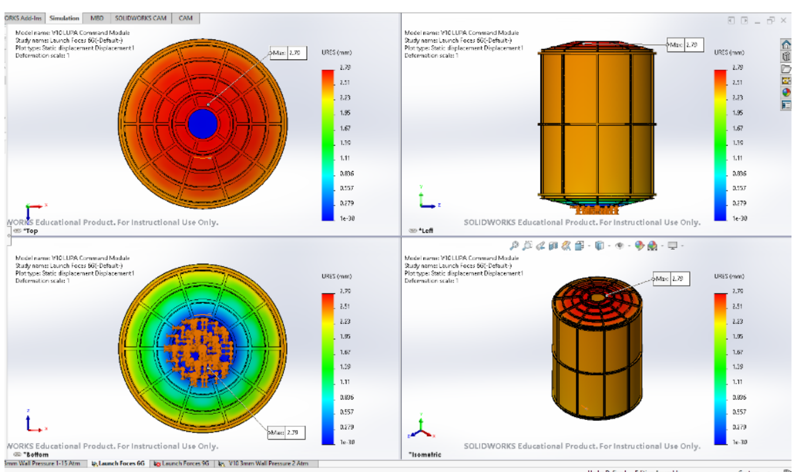


Figure 57: V10 LUPA Command Module 3mm Wall Launch Forces 6G Displacement FEA

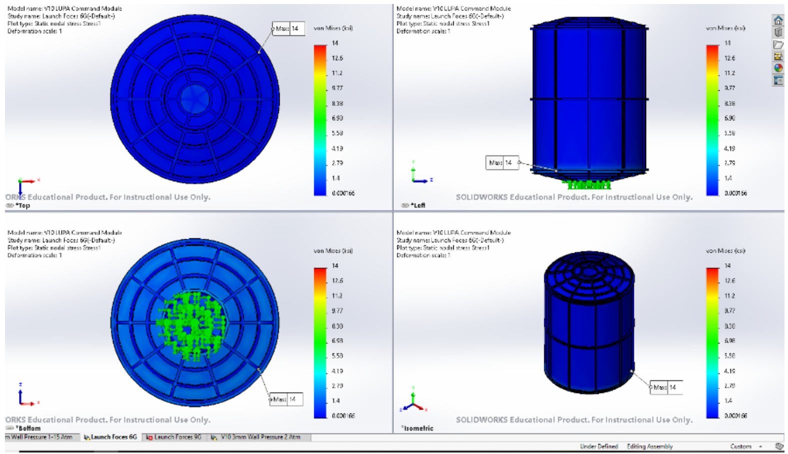


Figure 58: V10 LUPA Command Module 3mm Wall Launch Forces 6G Stress FEA

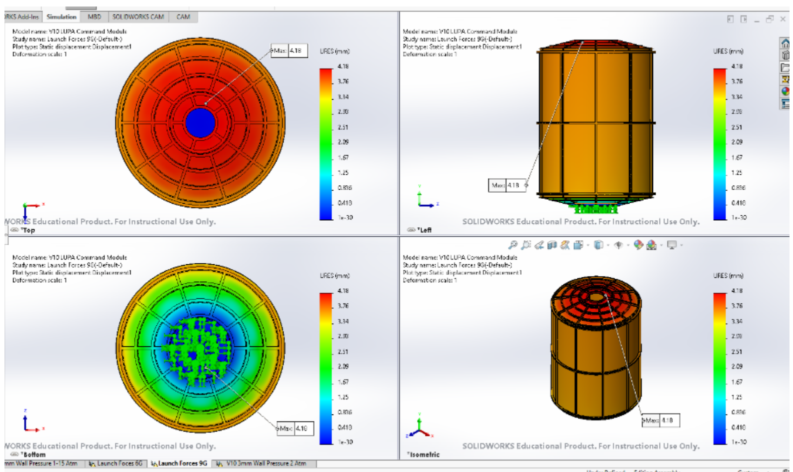


Figure 59: V10 LUPA Command Module 3mm Wall Launch Forces 9G Displacement FEA

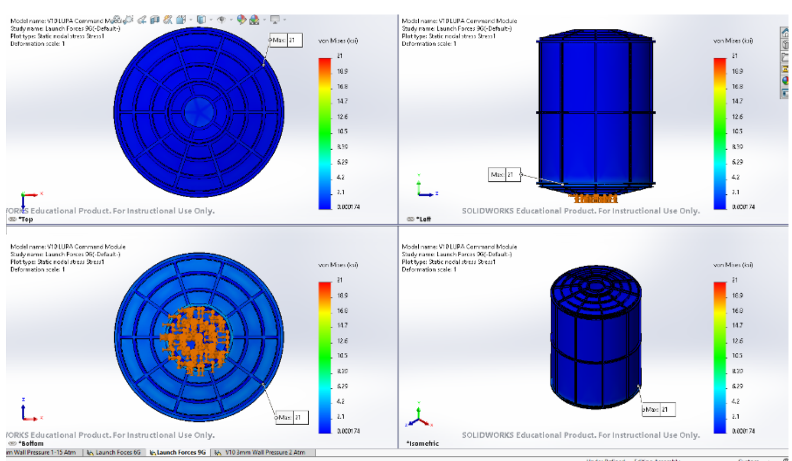


Figure 60: V10 LUPA Command Module 3mm Wall Launch Forces 9G Stress FEA

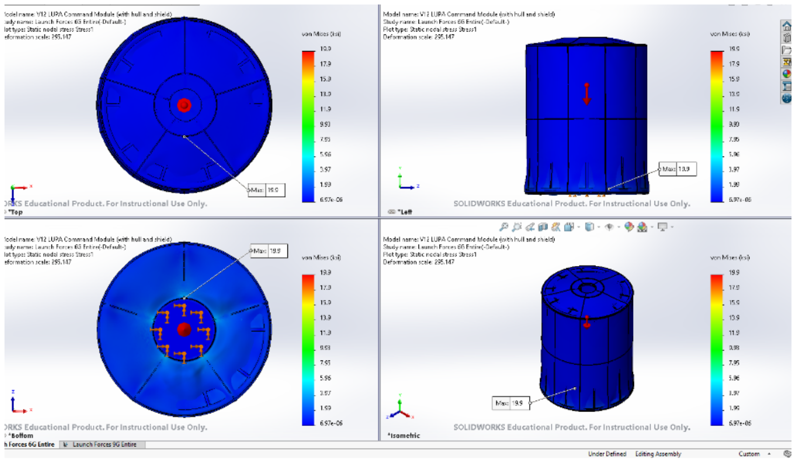


Figure 61: V12 LUPA Command Module 3mm Wall Everything Launch Forces 6G Stress FEA

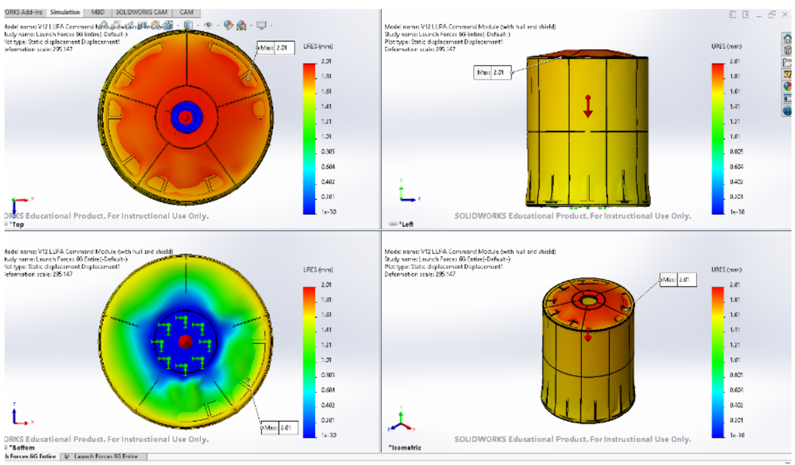


Figure 62: V12 LUPA Command Module 3mm Wall Everything Launch Forces 6G Displacement FEA

18 Appendix D Unorganized Handwritten Notes and Drawings

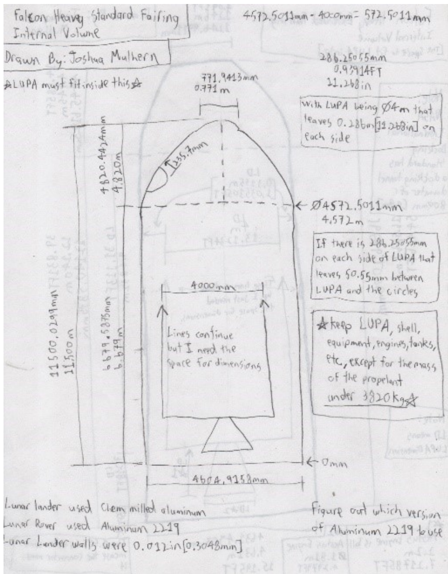


Figure 63: Hand Drawing Falcon Heavy Standard Fairings Internal Volume

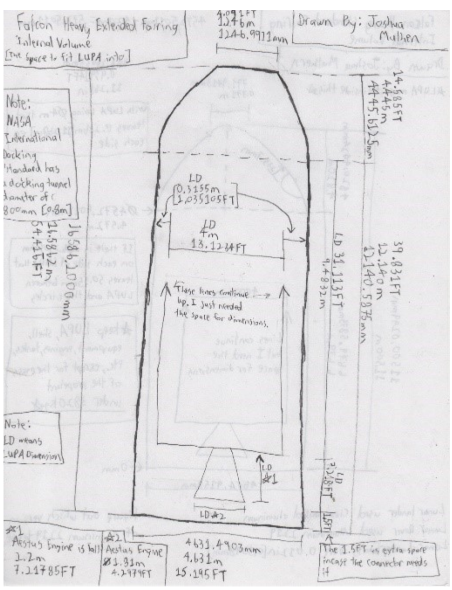


Figure 64: Hand Drawing Falcon Heavy Extended Fairings Internal Volume

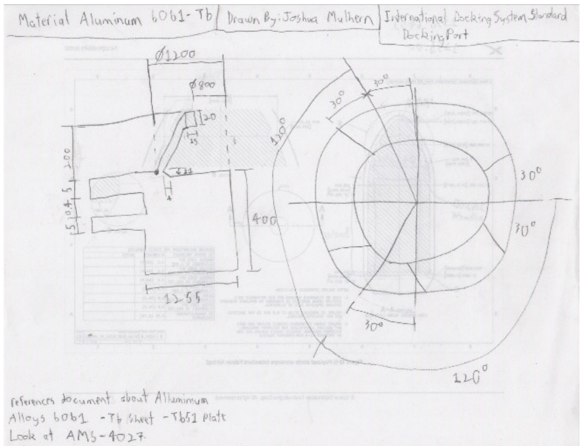
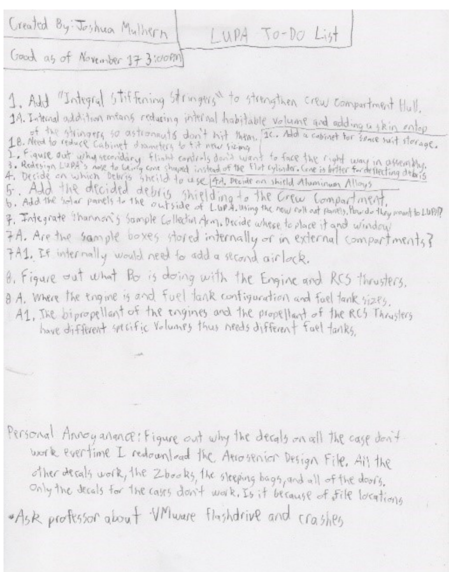


Figure 65: Hand Drawing International Docking System Standard Docking Port


Created By: Joshua Mulhern
LUPA To-Do List
Good as of November 17 3:00PM

1. Add "Integral Stiffening Stringers" to strengthen Crew Compartment Hull.
1A. Internal addition means reducing internal habitable volume and adding a skin ontop of the stringers so astronauts don't hit them.
1C. Add a cabinet for space suit storage.
1B. Need to reduce Cabinet diameters to fit new sizing
2. Figure out why secondary flight controls don't want to face the right way in assembly.
3. Redesign LUPA's nose to being cone shaped instead of the flat cylinder. Cone is better for deflecting debris
4. Decide on which Debris sheild to use
4A. Decide on shield Aluminum Alloys
5. Add the decided debris shielding to the Crew Compartment.
6. Add the solar panels to the outside of LUPA. Using the new roll out panels, How do they mount to LUPA?
7. Integrate Shannon's Sample Collection Arm. Decide where to place it and window
7A. Are the sample boxes stored internally or in external compartments?
7A1. If internally would need to add a second airlock.
8. Figure out what Bo is doing with the Engine and RCS thrusters.
8A. Where the engine is and fuel tank configuration and fuel tank sizes.
A1. The bipropellant of the engines and the propellant of the RCS Thrusters have different specific Volumes thus needs different fuel tanks.

Personal Annoyanance: Figure out why the decals on all the case don't work evertime I redownload the Aerosenior Design File. All the other decals work, the Zbooks, the sleeping bags, and all of the doors. Only the decals for the cases don't work. Is it because of file locations

*Ask professor about VMware flashdrive and crashes

Figure 66: Hand Drawing LUPA To-do list as of November 17th

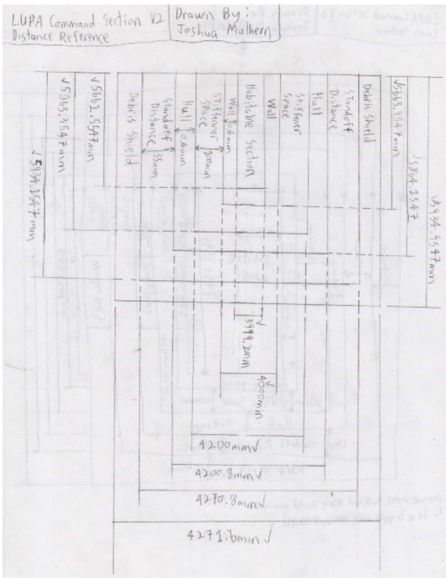


Figure 67: Hand Drawing LUPA Command Section V2 Distance Reference

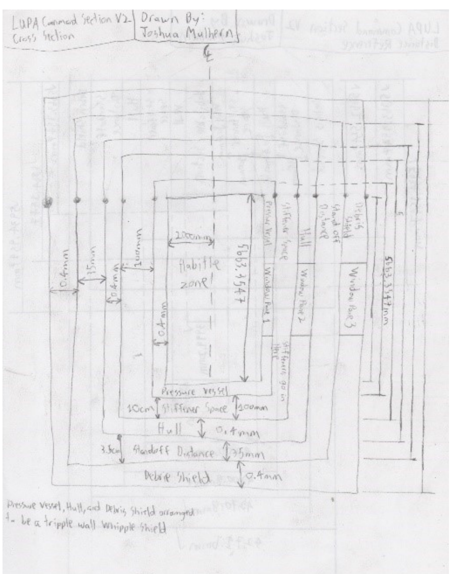


Figure 68: Hand Drawing LUPA Command Section V2 Cross-Section

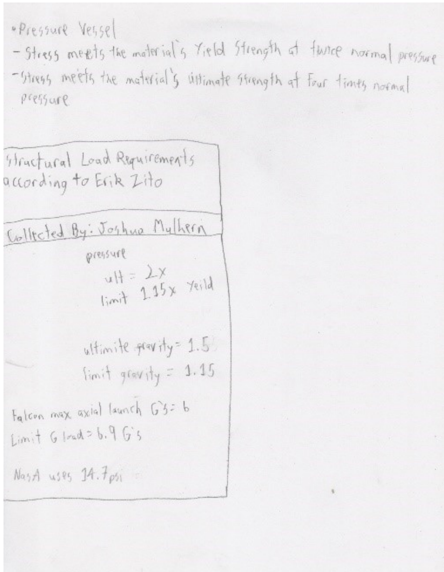


Figure 69: Hand Drawing Structural load requirements according to Eric Zito

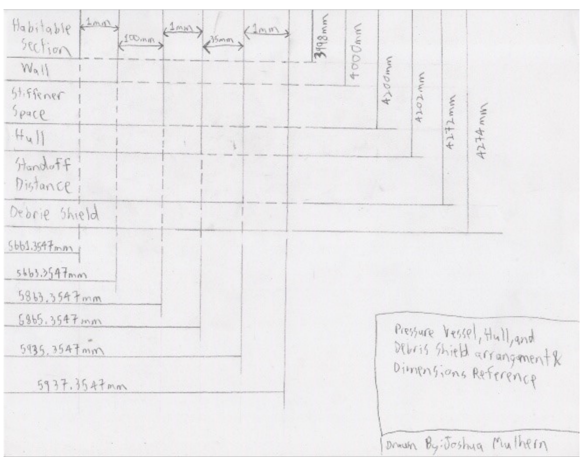


Figure 70: Hand Drawing Pressure vessel, hull, debris shield arrangement and dimension reference

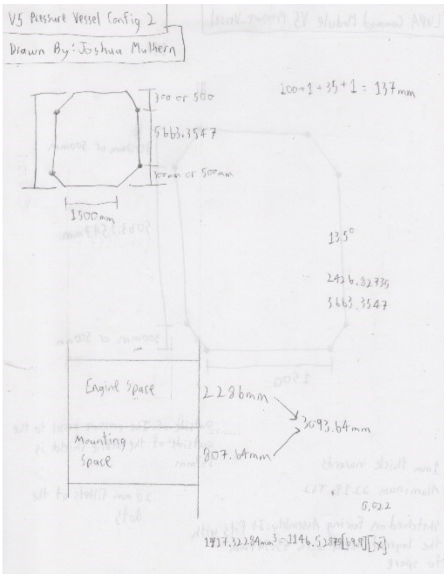


Figure 71: Hand Drawing V5 Pressure Vessel Congfig 2

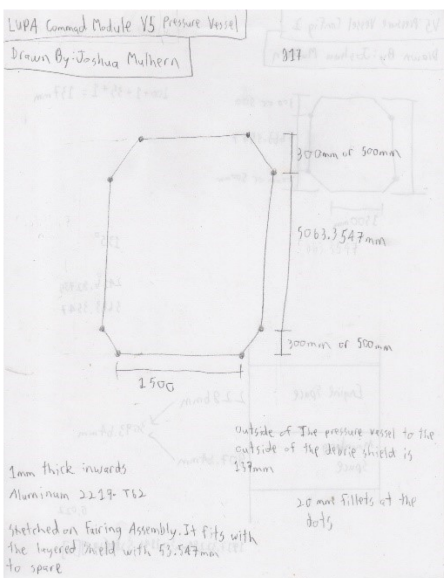


Figure 72: Hand Drawing LUPA Command Module V5 pressure vessel

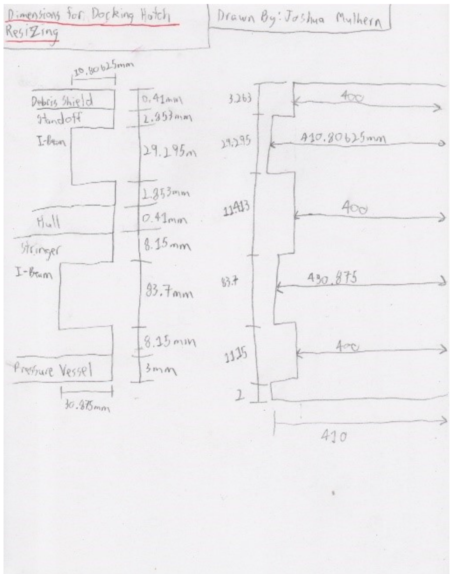


Figure 73: Hand Drawing Dimension for docking hatch resizing

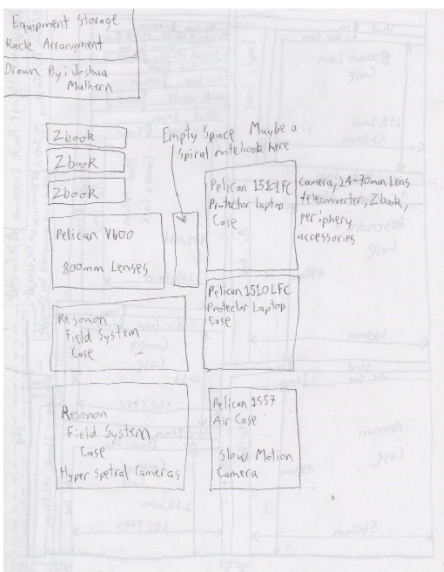


Figure 74: Hand Drawing Equipment storage rack arrangement

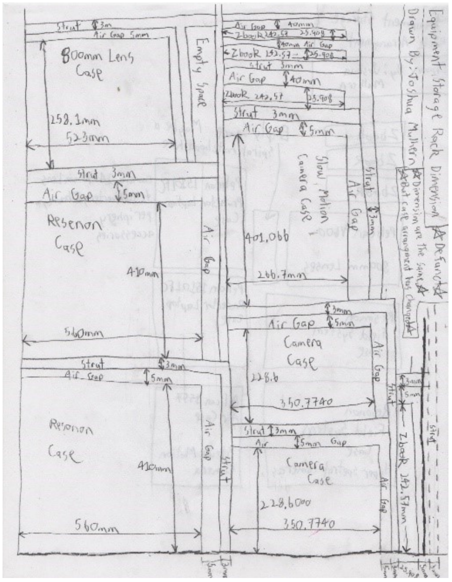


Figure 75: Hand Drawing Equipment storage rack dimensions

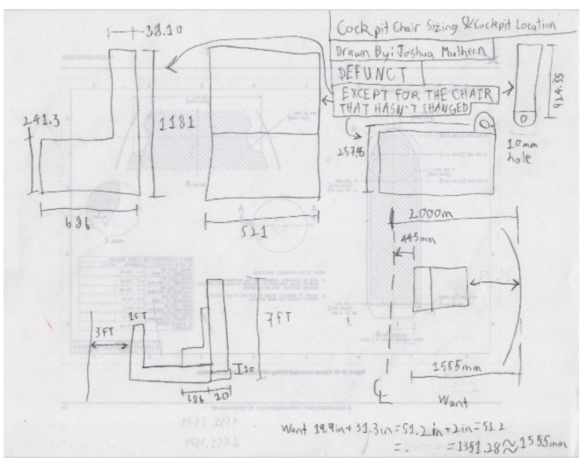


Figure 76: Hand Drawing Cockpit Chair sizing and cockpit location

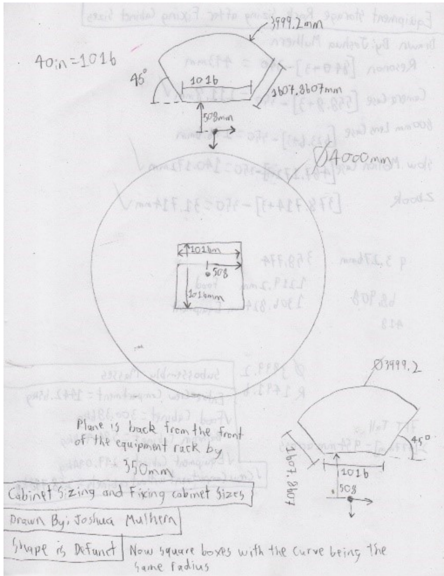


Figure 77: Hand Drawing Cabinet sizing and fixing cabinet sizes

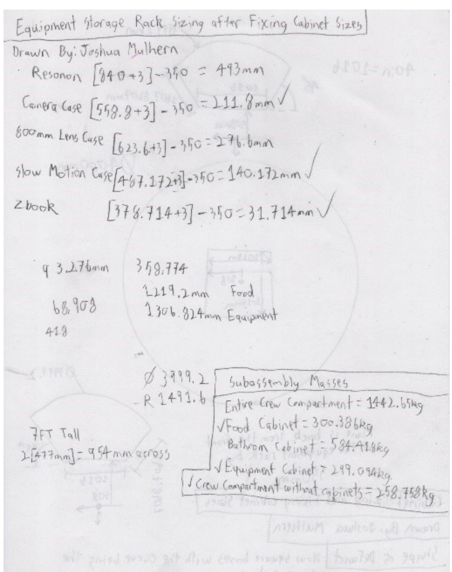


Figure 78: Hand Drawing Equipment storage rack sizing after fixing cabinet sizing

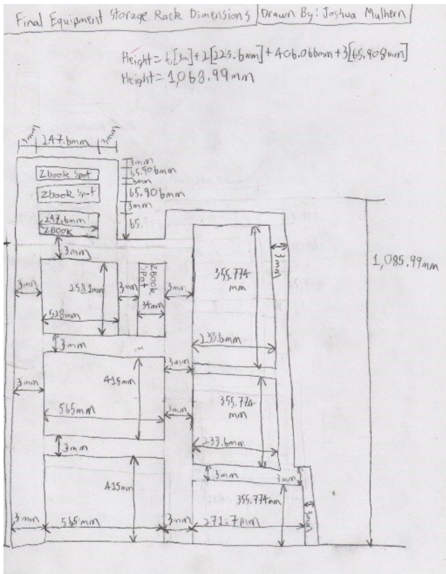


Figure 79: Hand Drawing Final equipment rack dimensions

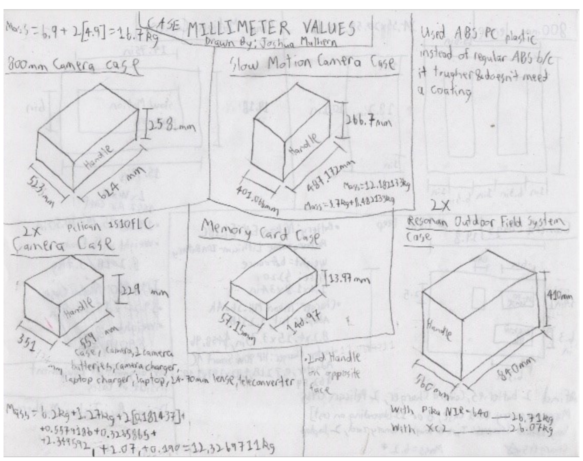


Figure 80: Hand Drawing Case millimeter values

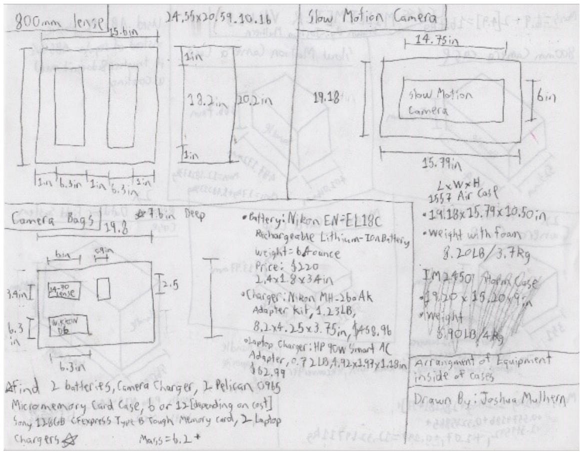


Figure 81: Hand Drawing Arrangment of equipment inside of cases

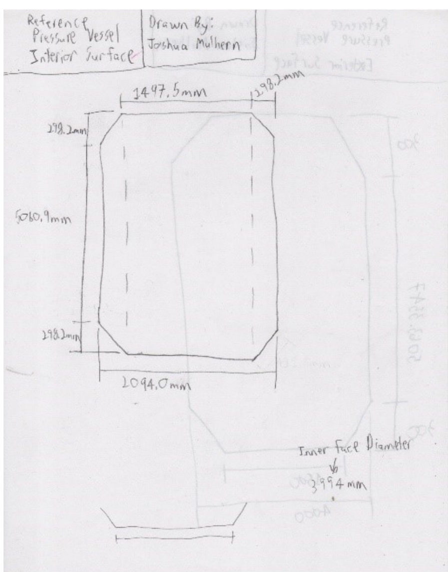


Figure 82: Hand Drawing Reference pressure vessel interior surface

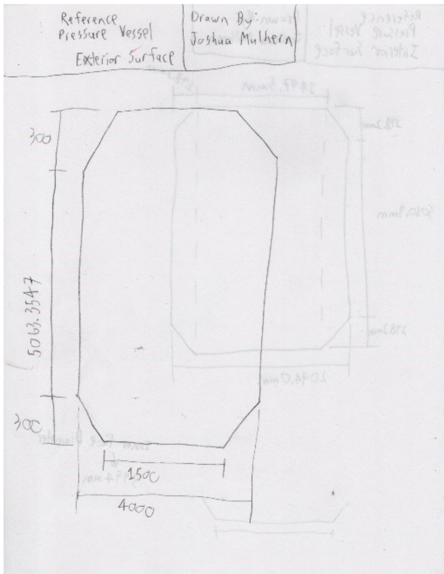


Figure 83: Hand Drawing Reference pressure vessel exterior surface

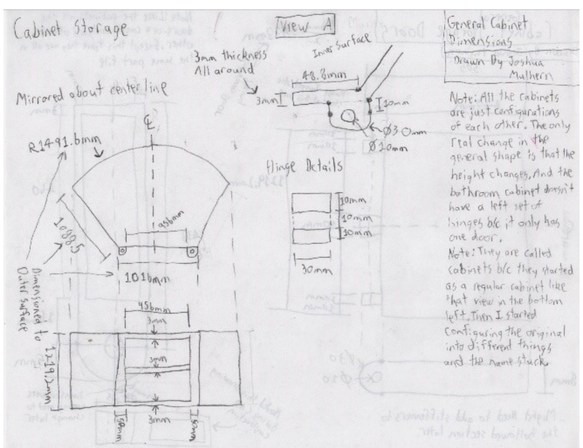


Figure 84: Hand Drawing General cabinet dimensions

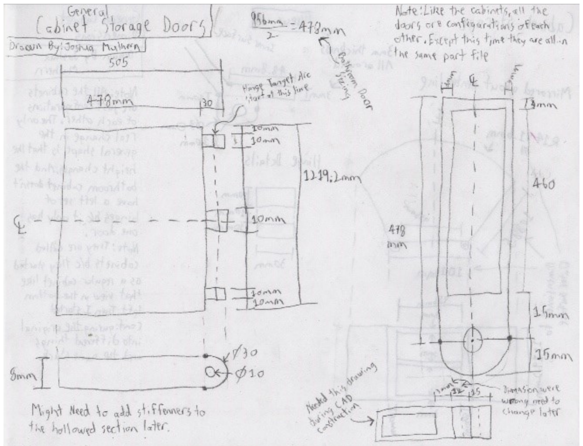


Figure 85: Hand Drawing General cabinet storage doors

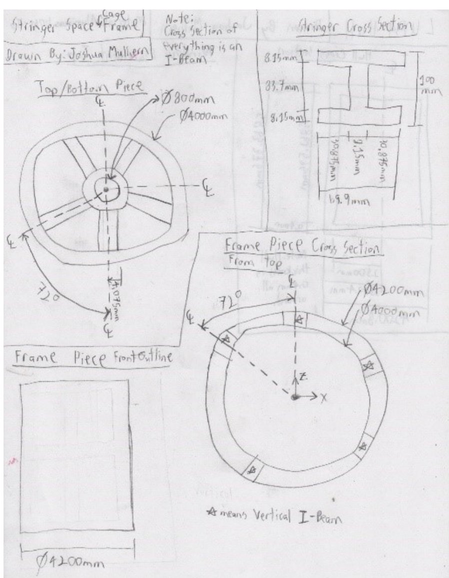


Figure 86: Hand Drawing Stringer space cage frame

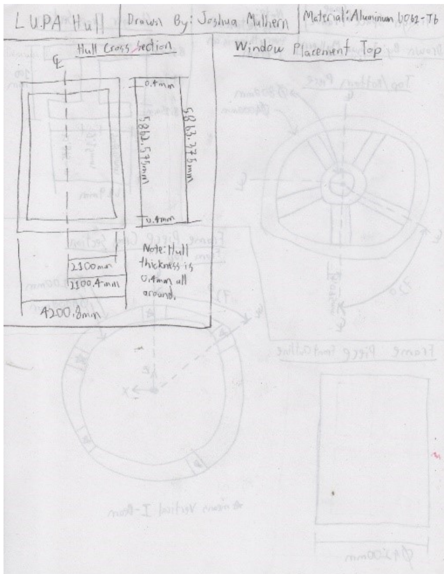


Figure 87: Hand Drawing LUPA hull

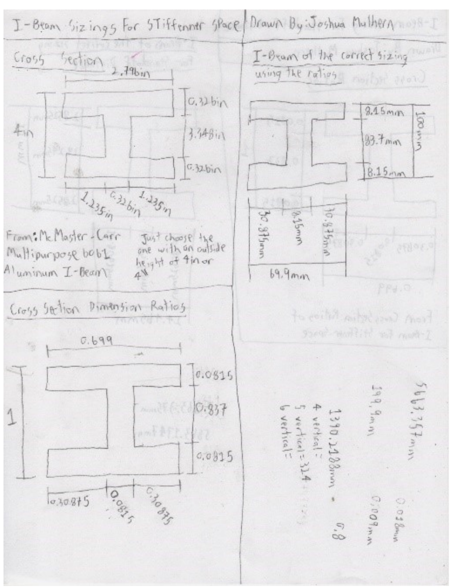


Figure 88: Hand Drawing I-beam sizing for stringer space

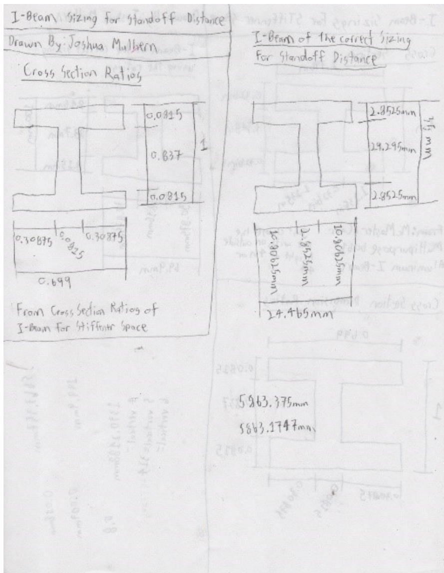


Figure 89: Hand Drawing I-beam sizing for standoff distance

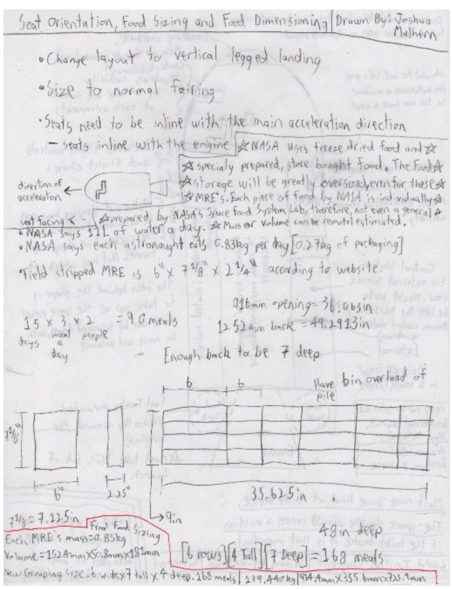


Figure 90: Hand Drawing Seat orientation, food sizing, and food dimensioning

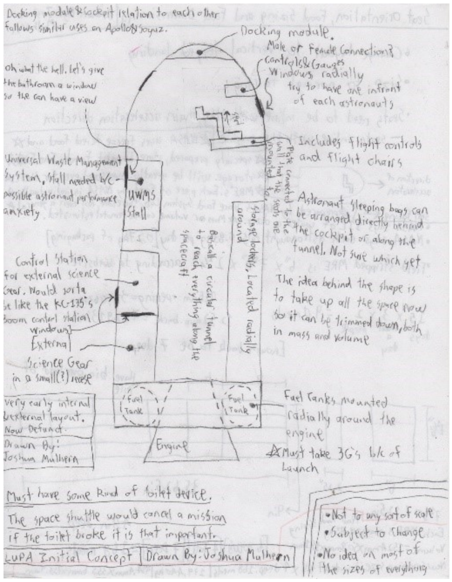


Figure 91: Hand Drawing LUPA initial concept

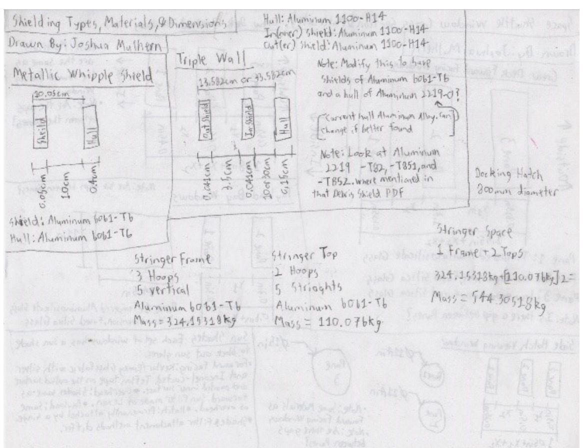


Figure 92: Hand Drawing Shielding types, materials, and dimensions

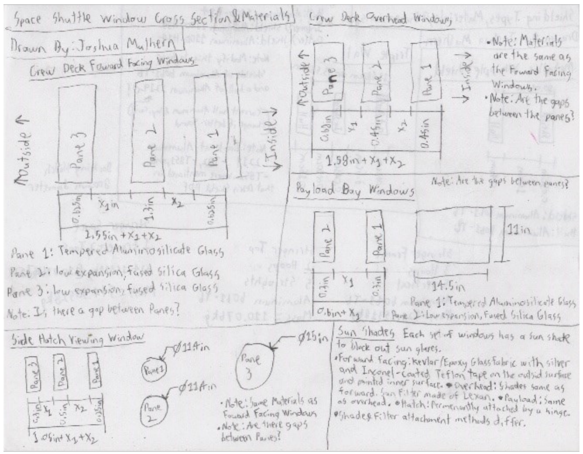


Figure 93: Hand Drawing Space shuttle window cross-section and materials

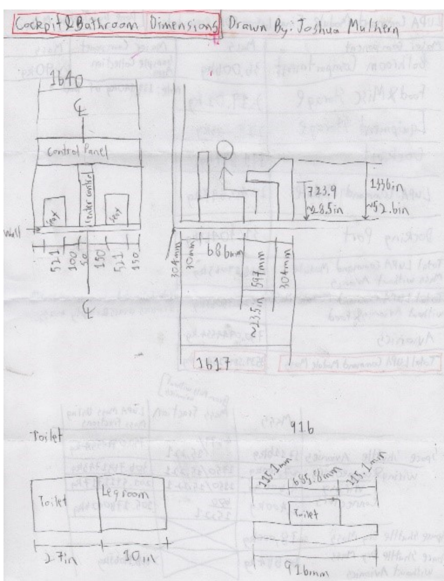
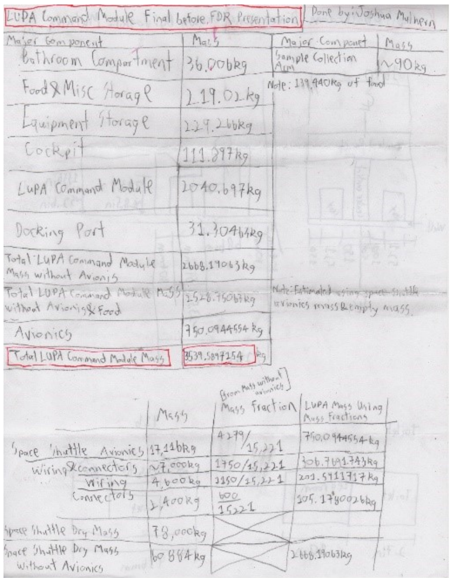


Figure 94: Hand Drawing Cockpit and bathroom dimensions



LUPA Command Module Final before FDR Presentation

Done by: Joshua Mulhern

Major Component	Mass
Bathroom Compartment	36.006kg
Food & Misc Storage	219.02kg
Equipment Storage	229.266kg
Cockpit	111.897kg
LUPA Command Module	2040.697kg
Docking Port	31.30464kg
Total LUPA Command Module Mass without Avionis	2668.19063kg
Total LUPA Command Module Mass without Avionis & Food	2528.75063kg
Avionics	750.0944554 kg
Total LUPA Command Module Mass	3539.5897254 kg

Major Component	Mass
Sample Collection Arm	~90kg

Note: 139.440kg of food

Note: Estimated using space shuttle avionics mass & empty mass.

	Mass	Mass Fraction [from Mass without avionics]	LUPA Mass Using Mass Fractions
Space Shuttle Avionics	17,116kg	4279/15,221	750.0944554kg
Wiring & connectors	~7,000kg	1750/15,221	306.7691743kg
Wiring	4,600kg	1150/15,221	201.5911717kg
Connectors	2,400kg	600/15221	105.1780026kg
Space Shuttle Dry Mass	78,000kg		
Space Shuttle Dry Mass without Avionics	60884kg		2668.19063kg

Figure 95: Hand Drawing LUPA Command Module final mass before FDR presentation

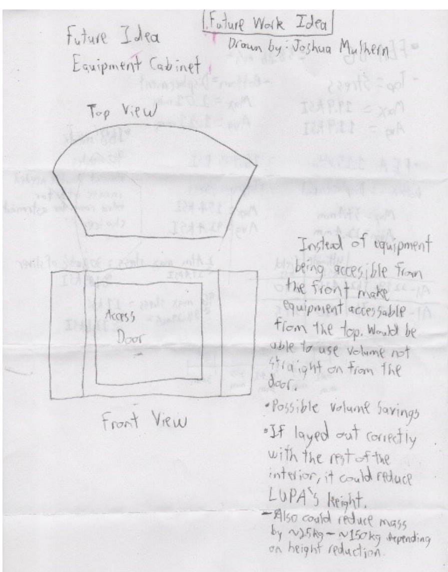


Figure 96: Hand Drawing Future work idea

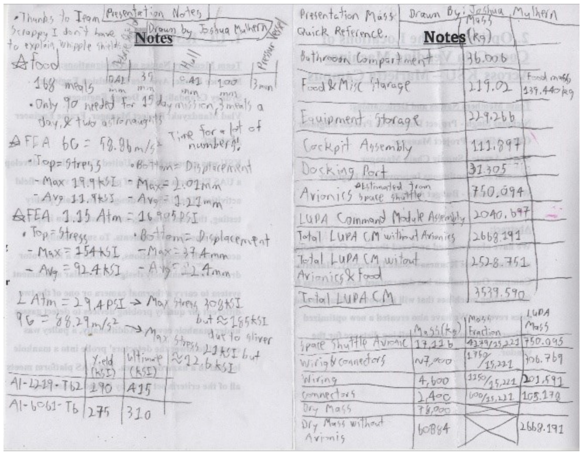


Figure 97: Hand Drawing Presentation notes and presentation mass quick reference